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Contributors to This Issue

Dr. Wolfgang B. Klemperer is chief of the Research Division of the Goodyear-Zeppelin Corporation and is perhaps better known to the members of the Soaring Society as the originator of the modern high performance sailplane. He was for some time assistant to Professor Von Kármán at Aachen, where in 1924 he took his doctorate in engineering. Later he was in charge of the wind tunnel of the German Zeppelin Company. During the War he was in the Austrian Air Service as pilot, instructor and designer.

Alfred V. de Forest is a consulting metallurgical engineer and inventor of the scratch extensometer and magnetic powder method of examination for defects in steel parts.

S. del Proposto is a member of the staff of the Aerodynamical Institute at Aachen and submitted the paper printed in this issue at the request of Professor Hopf of Aachen.

Professor Edward A. Stalker is in charge of Aeronautical Engineering at the University of Michigan, from which he graduated in 1919. He was associated with W. B. Stout as designer in 1920–22.

H. C. Richardson, Captain, Construction Corps, U. S. Navy (Retired), is best known as a pioneer designer and pilot of flying boats. He participated in the Navy's N. C. trans-Atlantic adventure in 1919.

H. C. Houghton, a Research Associate in the Department of Electrical Engineering at the Massachusetts Institute of Technology, has been engaged for several years in the study of fog, which is here reported.

C. Fayette Taylor is Professor of Automotive Engineering at the Massachusetts Institute of Technology. A graduate of Yale, he served in both the Army and the Navy Air Services during the War, was in charge of the Power Plant Laboratory at McCook Field, 1920–23, and was Assistant Chief Engineer of the Wright Aeronautical Corporation, 1923–26.

Harold Caminez is engineer in charge of aircraft engine development with the Allison Engineering Company. After graduating from Cornell, he was with the Army Air Service Engineering Division and later became Chief Engineer of the Fairchild-Caminez Engine Company.

R. T. Sauerwein is a graduate of Johns Hopkins and is now a Research Assistant at the Guggenheim Aeronautical Laboratory of the Massachusetts Institute of Technology.

R. Cazaud is a member of the technical staff of the French Air Ministry and submitted the paper printed in this issue at the request of Inspector General Sabatier, Director of the Service des Fabrications de l'Aeronautique and a member of the I. Ae. S.

JOURNAL OF THE AERONAUTICAL SCIENCES

Volume 1 JULY, 1934 Number 3

Research on Fog at the Round Hill Research Station of the Massachusetts Institute of Technology

H. G. HOUGHTON, *Massachusetts Institute of Technology**

(Received April 26, 1934)

FOR the past four years a research on fog has been conducted at the Round Hill, South Dartmouth, Massachusetts estate of Colonel E. H. R. Green by members of the staff of the Massachusetts Institute of Technology. Round Hill is located on the west shore of Buzzards Bay, about eight miles south of New Bedford and frequent natural fogs make this an ideal spot for this work.

At the time of its inception the general purpose of this research program was the investigation and development of aids to navigation in fog. Because of the use of lights as guides in clear as well as in foggy weather the first problem concerned the transmission of light through fog. At that time little quantitative data were available on the subject. Utterback[1] had made some measurements in an artificial fog chamber, as had Anderson,[2] and Granath and Hulburt[3] had made measurements in natural fogs. The experimental conditions and the results were so different that it was difficult to arrive at a definite conclusion and it was therefore decided to undertake an experimental study of the transmission of visible light through artificial fog.[4] The fog was produced in a chamber about ten feet long by the condensation of low pressure steam and transmission data were obtained for eight bands of radiation in the visible spectrum at several fog densities. The bands were obtained from an incandescent lamp by means of glass and gelatine filters and were from 0.03 to 0.05μ in width. The solid curve of Fig. 1 is typical of the transmission curves which were obtained. The definite maximum at 0.490μ had not been anticipated

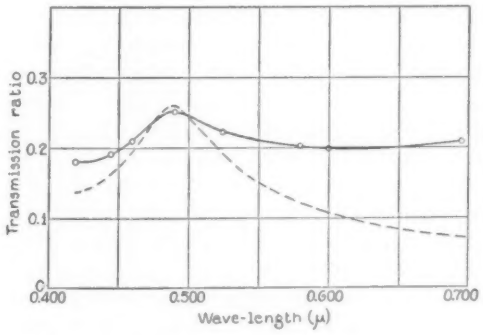

FIG. 1. Curves for the transmission of light through an artificial fog formed by the condensation of steam. The solid curve represents the observed data and the dotted curve the computed values.

* Department of Electrical Engineering, Round Hill Research Division.

[1] C. L. Utterback, Trans. I. E. S. **14**, 133 (1919).

[2] S. H. Anderson, Aviation 28, No. 19, May 10, 1930.

[3] Granath and Hulburt, Phys. Rev. **34**, 140 (1929).

[4] Houghton, Phys. Rev. **38**, 152 (1931).

from the results of other investigators and its presence prompted a theoretical investigation of the transmission of light through fog.[5] The development of the theoretical expression was based on the electromagnetic theory of light scattering by dielectric spheres which had been developed by Mie and Debye for somewhat different applications. The expression for the transmission was found to be

$$E/E_0 = e^{-2\pi n a^2 Z K}, \qquad (1)$$

where E_0 = initial light intensity.
E = light intensity after it has traversed Z cm of fog.
n = number of fog particles in one cc.
a = radius of fog particles in cm.
K = scattering coefficient which is plotted in Fig. 2.

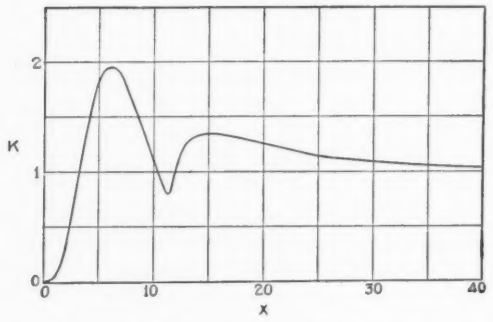

FIG. 2. Curve of the coefficient of scattering as a function of x where $x = 2\pi a/\lambda$.

An inspection of Fig. 2 shows that a transmission maximum would be expected at a value of λ which would make x equal to 11.2. If the observed maximum at 0.490μ corresponds to this minimum, $2a$ the particle diameter, must be $(11.2 \times 0.490)/\pi = 1.75\mu$. At the time this work was being done no accurate means for the measurement of the particle diameter was available but approximate microscopic measurements indicated that the drop diameter was $2.5 \pm 0.5\mu$ which agrees as well as can be expected with the computed value given above. The dotted curve of Fig. 1 was computed from (1) with $2a = 1.75\mu$ and with a value of n such that the ordinate at 0.490μ was about the same as that of the experimental curve. The theoretical curve is sharper than the experimental curve because it was computed on the assumption that the particles were all of the same size whereas there was actually a considerable range of particle sizes in the fog.

The work on the transmission of light indicated the need for data on the instantaneous size distribution and number of fog particles. A large number of determinations of fog particle size had been made by Kohler[6] largely by measuring the angle subtended by coronas in natural fogs and clouds. Werner,[7] Barus[8] and others had also used the same method both in natural and artificial fogs but such measurements give only the average or predominant particle size and not the instantaneous size distribution which was needed. It was also desirable to have apparatus available to measure the size distribution and number of the particles simultaneously with light transmission measurements so that the theoretical transmission formula could be adequately tested. It was therefore decided to undertake the development of the necessary apparatus for making these determinations. The first instrument which was constructed was a special microscope for measuring the diameter of individual fog drops.[9] The drops were collected on a greased glass slide which was viewed with the aid of dark-field illumination. Size measurements were made either directly, by means of an ocular micrometer, or by the subsequent measurement of photomicrographs. Flattening of the drops on the slide was corrected for by an experimentally determined coefficient which was found to be independent of the drop size within the range of sizes encountered and dependent only on the kind of grease used on the slide. Typical size distribution curves for natural fogs at Round Hill are given in Fig. 3. The group of three curves taken at half-hour intervals is unusual in that the predominant particle size changed so rapidly but it illustrates an extremely interesting, but as yet unexplained, phenomenon. The maxima of these three curves, as well as those of all of the curves which have been ob-

[5] Stratton and Houghton, Phys. Rev. **38**, 159 (1931).

[6] Hilding Kohler, Met. Zeits. **38**, 359 (1921); **40**, 257 (1923); **42**, 137 (1925), and Meddelanden fran Stratens Meteorologiska-Hydrografiska Anstalt, Stockholm (1925).

[7] O. Werner, Ann. d. Physik **70**, 375, 480 (1923).

[8] C. Barus, Phil. Mag. **4**, 24 (1902) and Am. J. Sci. **175**, 224 (1908).

[9] Houghton, Physics **2**, 467 (1932).

tained over a period of three years, are integral multiples of 3.1μ. It has been definitely established that this is not due to peculiarities inherent to the method of measurement and it is thought that it must be intimately connected with the nature of the nuclei of condensation and the mechanism of fog formation. These results are apparently not in agreement with Kohler's findings. As a result of numerous corona measurements, principally at Haldde at an altitude of 600 meters, he concluded that there was an integral relation between the average particle volumes rather than the diameters. More accurate and complete data on fogs at Round Hill are required to confirm their apparent fundamental difference from the fogs and clouds observed by Kohler. Preliminary development work has been started on apparatus which it is hoped will enable the particle size distribution to be determined much more accurately.

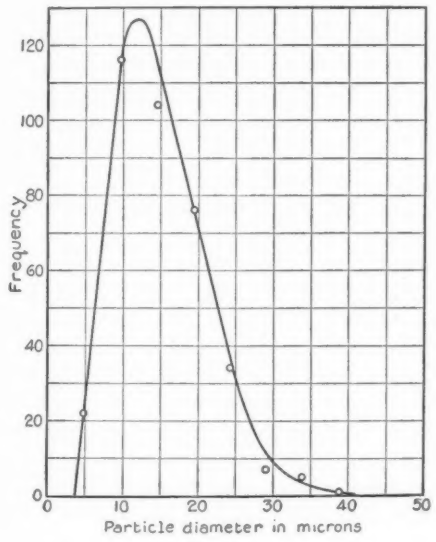

FIG. 3a.

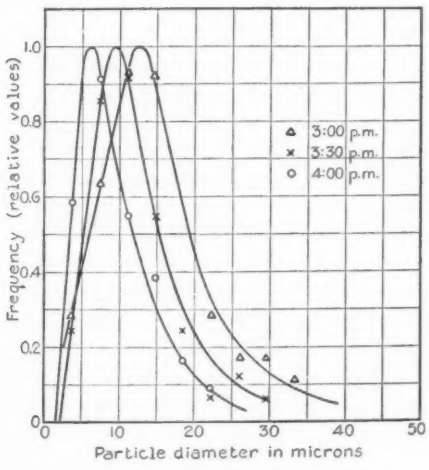

FIG. 3b.

FIG. 3. Size distribution curves for natural fogs. The group of three curves was taken at half-hour intervals in the same fog.

During a given fog it has been found that the particle size distribution tends to remain practically constant so that the group of three curves in Fig. 3 is unusual and apparently indicates a definite change in the characteristics of the fog. The most common values for the predominant particle diameter at Round Hill are 12.4 and 18.6μ. It has not yet been possible to correlate the particle size with other pertinent characteristics of the fog.

Although preliminary work has been started an instrument for the direct determination of the number of fog particles in a unit volume of foggy air has not yet been constructed. For the present this figure is obtained by computation from the amount of liquid water in a unit volume and the size distribution of the fog particles. The liquid water content has been obtained by other workers by measuring the total water content, which includes both the liquid water and the water vapor, and subtracting the weight of the water vapor, which was obtained on the assumption that the air was saturated at the ambient temperature. This method is inaccurate because of the fact that the weight of the water vapor is much greater than the weight of the liquid water. A special fog calorimeter has been developed which measures the liquid water content by measuring the heat required to evaporate the liquid water. This method not only gives the weight of liquid water directly but also quite accurately because of the relatively large value of the latent heat of evaporation. Before this instrument could be constructed it was necessary to have information on the time required for the complete evaporation of fog drops as a function of the temperature and relative hu-

midity. This problem has been treated both theoretically and experimentally in a recent paper[10] and it was found that the time in seconds required for the complete evaporation of a drop of initial diameter A centimeters was given by

$$T=A^2/8k(D_0-D), \tag{2}$$

where k = the coefficient of diffusion (about 0.25 cm²/sec.)
D_0 = water vapor density at the surface of the drop.
D = water vapor density in the atmosphere at a distance from the drop.
(D and D_0 are densities with respect to liquid water).

The drop temperature is reduced below the ambient temperature because of the heat of evaporation and D_0 is the saturation vapor density corresponding to the true drop temperature. The drop temperature may be computed approximately by means of the following expression

$$e-e'=-12.3\times10^{-4}B(t-t'), \tag{3}$$

where t = air temperature in °C,
t' = drop temperature in °C,
e = vapor pressure corresponding to t,
e' = vapor pressure corresponding to t',
B = barometric pressure measured in the same units as e and e'.

The fog calorimeter is now in operation but the details of its construction and operation have not yet been published. It consists essentially of a well-insulated vertical pipe about eight feet long with an electric heater located near the intake at the top and a constant-speed fan at the bottom to provide a uniform airflow through the instrument. Provisions are made for measuring the inlet and outlet temperatures and the power input to the heater. The liquid water content may then be obtained from the temperature rise, the power input to the heater and the airflow which is a constant.

In order to determine the actual electrical conductivity of the water of which fog drops are composed, apparatus has been constructed for collecting samples of fog water during natural fogs. This apparatus consists essentially of fine mesh nickel screens of considerable area, the plane of which is perpendicular to the wind direction. A considerable amount of fog water collects on the screens and runs down to the bottom where it is caught by a trough and conveyed to a bottle. Measurements of electrical conductivity have been made on such samples in the usual manner and it has been found that the assumption of a negligible conductivity which had been made in connection with the light transmission theory was justified. These results cannot be expected to hold for fogs in or near large centers of population where the products of combustion act as nuclei of condensation, thus greatly increasing the conductivity. The work on the conductivity of fog water has been extended to include chemical analyses in the hope of learning something of the nature and size of the nuclei of condensation, but definite conclusions have not yet been arrived at.

Apparatus having been developed for the measurement of particle size and number as outlined above, it was decided to make simultaneous measurements of light transmission and particle size and number in natural fogs, in an attempt to check the theoretical light transmission formula in a more satisfactory manner. Accordingly, the necessary instruments for the measurement of the transmission of light from about 0.35 to 1.0μ over a path length of one hundred and fifty feet have been set up permanently out of doors. Computations have indicated that the transmission curves for natural fogs will be practically flat throughout the range over which these measurements will be made, but the theoretical formula can be checked by computing the actual magnitude of the transmission ratio from the measured drop size and number.

The program for the future calls for continued measurements of the physical and chemical properties of such natural fogs as occur and the development of new and better apparatus for these measurements. Sufficient light transmission data will be obtained to definitely prove or disprove the theoretical formula and the investigation of the nature of the condensation nuclei will be continued and extended because this is evidently of fundamental importance.

[10] Houghton, Physics **4**, 419 (1933).

The Stalling of Airships

W. B. KLEMPERER, *Goodyear-Zeppelin Corporation*
(Received November 17, 1933)

A DIRIGIBLE airship can be "stalled." This term is derived from airplane parlance. It signifies a peculiar flight condition in which the efficiency of the aircraft as a load carrier is greatly reduced and the response to the controls is different from the usual response in normal flight. Stalled flight may under certain circumstances be a temporary condition, which quickly leads either to recovery of normal flight or to catastrophic loss of control. Under other circumstances stalled flight may be a stable enough form of flight to be maintained permanently or deliberately. But the elevator controls then act as though reversed.

Stalled flight of airships and of airplanes are not altogether identical phenomena but they are quite closely related, inasmuch as stalling is a purely aerodynamical phenomenon and concerns the dynamic lift and drag of the craft. It has nothing to do with the aerostatic buoyancy of a gas inflated airship. However, an airship flying "light" or "heavy" (i.e., carrying less or more load than its gas lifts) derives aerodynamic lift from its motion through the encountered air in much the same way as an airplane. As such it may be subject to stalling. An airship in buoyancy equilibrium cannot really be stalled.

The stall is a characteristic of those aerodynamical devices which produce a reaction force normal to their direction of motion through the fluid (air), such as airplane wings, propeller and helicopter blades, fan and blower blades, etc., where this normal force is a function not only of the velocity head[1] of the motion but also of the angle of attack. In the efficient range of angles of attack, the "lift coefficient" (i.e., the cross-wind force per unit velocity head and area) increases with the angle of attack. The drag is a minimum for the angle of attack of zero lift for symmetrical airfoils (or of some small lift coefficient for curved or unsymmetrical airfoils). It increases with any change of the angle of attack therefrom. In airplane flight this regime of angles of attack is "stable," because, if equilibrium of forces is attained at a certain angle of attack, any deviation from it will elicit free forces acting towards restorage of conditions. For instance, should the lift be accidentally decreased (by a gust or an elevator movement, etc.) then the surplus weight will make the ship lose altitude and this slopes the path of flight downward, increasing the angle of attack against the ship, thus increasing the lift; at the same time the decreased drag will accelerate the ship and the increased speed will also cause the lift to increase, together with the drag, until equilibrium in both directions is reestablished. The reestablishment of lift equilibrium proceeds faster than that of drag. However, the larger the angle of attack, the less completely can the air follow the wing or blade curve on the suction (leeward) side, and the smaller the rate of increase of lift and the larger the rate of increase of drag. There usually is a certain angle of attack for which the lift is largest. At larger angles the lift actually drops and the drag increases rapidly. In this regime airplane flight is unstable; any drop of lift that lets the flight path incline downwards and thus increases the angle of attack entails another drop of lift; increase of drag thus induces more rapid loss of height, again an increase of angle of attack and this repeats the cycle of consequences. This sort of stall is a normal maneuver in a good landing. It is performed so close to the ground that the wheels touch the ground immediately after attaining the angle of attack of maximum lift coefficient. It is attained at the slowest flying speed.

An airplane having "flat" maximum lift characteristics may be flown stalled by surplus

[1] Velocity head = half air density × velocity^2.

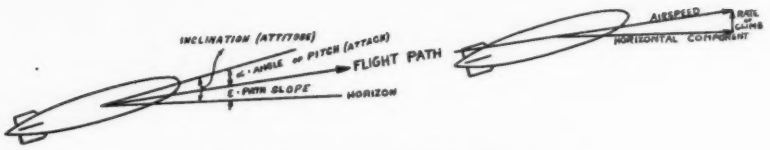

FIG. 1. Nomenclature of flight geometry.

power (either motor power or gravity power in a steep descent) and may easily be "recovered." An airplane having "vicious" stall characteristics, i.e., one where the air flow suddenly detaches from the upper wing curvature upon reaching the stalling angle and where the lift then suddenly drops by a large amount and drag suddenly increases by a large amount, requires a considerable gain in flying speed before recovery can be effected. Often the angle of attack of stalling is then larger than that of recovery (hysteresis). Stalling is in general the more pronounced the better the wing for high lift. A poor or low lift wing may have a flat stalling range.

Whereas the stalling of airplanes is mostly governed by a sudden loss of lift from detachment of air flow, the stalling of airships is more a phenomenon of want of tractive power or of control.

Airships are relatively poor wings and consequently their stalling characteristics are flat and not vicious. Airships are usually built to fly near buoyancy equilibrium, i.e., near zero aerodynamic lift. They are essentially symmetrical and as such offer least drag at zero angle of attack. The angle of attack is the angle between the ship's axis and its path in air. In the vertical (keel) plane it is called the angle of pitch. It must not be confused with the ship's inclination (attitude) against the horizon. The two are identical only when the ship maintains its altitude. However, when it is rising or descending then the angle of pitch (attack) is the difference between the inclination angle (of attitude) of the ship's axis to the horizon and the slope of the ship's path against the horizon (Fig. 1).

When the ship is in equilibrium (and trim) the angle of pitch (attack) is necessarily zero (no dynamic lift) and the angles of attitude and of flight path slope are the same. In this condition there can be no stall, because equilibrium of forces is not disturbed by steeply climbing or descending. The weight W is always acting vertically down and the buoyancy B vertically up no matter how the ship is inclined. The drag D then always acts in the direction directly opposite the propeller thrust T. The two pairs of forces are individually in equilibrium although the pairs are then not at right angles to each other. The airship weighs nothing and is driven to where it points (Fig. 2).

Therefore an airship in equilibrium need not lose speed when steeply climbing nor pick up speed when descending. This is very impressive to airplane pilots when riding in an airship. Blimps have frequently been made to climb and descend at inclinations of 35° for demonstration purposes and large rigid airships have frequently passed through angles of inclination of 20° and more, bearing out the above contention.

Airships possessing static (metacentric) stability require elevator action in order to maintain them at an unusual angle of attitude. This elevator incidence produces additional drag and may perceptibly slow down the ship's speed. Where this is very pronounced, it may so limit

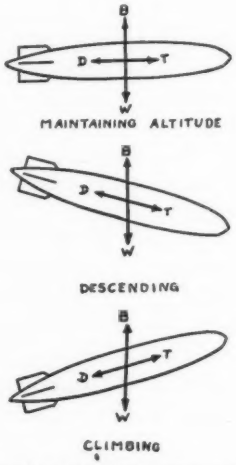

FIG. 2. Zero angle of attack (buoyancy equilibrium).

the maximum pitching moment to be derived from the elevator that the ship cannot be inclined beyond a certain angle of attitude and not for instance made to loop the loop. This limit is not a true stall, but merely a (safe) limitation of controllability. Nothing inadvertent happens upon reaching the limit and recovery toward normal attitude is under immediate and powerful control.

Airships statically seriously unstable or gravely out of trim, may theoretically "capsize," i.e., after loss of speed due to frantic elevator action the ship assumes an extreme angle of inclination from which it cannot recover without retrimming. Such incidents have never been reported except from completely disabled craft in the early days.

The longitudinal stability of airships in flight depends on three factors, *viz.*, the "static" stability due to the metacentric height, the "aerodynamic" stability due to the angle of attack, and the "dynamic" stability due to rotation. The static or metacentric stability is of importance at low speed but its relative importance diminishes as the other two factors increase with speed. The aerodynamic stability against angle of attack is slightly negative for many airships. The fact that they are yet controllable hinges upon the dynamic restoring moment derived as soon as free rotation builds up. Since the ship tends to turn about a "center of sway" in the region of her bow, the angle of attack increases locally from bow to stern so that at the stern it is about twice as large as at the center of buoyancy. Therefore in free rotation the fin is something like twice as powerful a stabilizer as in straight motion under uniform angle of attack. This difference is interpreted as the "dynamic" stability component. A ship with pronounced "aerodynamic" negative stability can be flown under good control when in equilibrium and the mechanism of the maintenance of attitude can be compared to the straight motion of a bicycle. However, a ship of such characteristics requires a "negative" average angle of elevator incidence in order to maintain a flight attitude under a constant angle of pitch with the ship heavy or light; i.e., when the ship is light, nose down, it requires up elevator to maintain this condition and the more so the more angle of pitch is to be maintained. The reason is that in flight with constant angle of pitch there is no rotation and no stabilizing benefit to be derived therefrom. The elevator angle is necessary to counterbalance the unstable aerodynamic moment. Some ships require such elevator angle only for moderate angles of pitch while for large angles of pitch the elevator angle required decreases again and aerodynamic stability may be reattained. The *Akron* and *Macon* belong to this class.

Captain P. Changeux, in *Dynamique du Dirigeable*,[2] investigated in great detail the behavior of an airship possessing a very much more pronounced aerodynamic unstability, i.e., a ship requiring large balancing elevator to trim when light or heavy. Evidently such a ship is considerably hampered in its maneuverability by its unsymmetrical available range of elevator. It has very little elevator range left for deliberately pitching it towards the direction of its dynamic lift and it has very ample elevator range for pitching it against it. That means such a ship when light reacts readily on any command to pitch it down but hesitatingly to pitch it up, and *vice versa*. Such a behavior is quite beneficial so long as the angle of attack is small because it assists the elevatorman in maintaining altitude. However, this same behavior becomes very obnoxious when a stall is approached because it is very conducive to accidental stalling.

A ship of this type of behavior may react in an unruly manner when ballast is dropped amidships or when gas is released amidships or distributed evenly about the center of buoyancy. For instance, if a ship was flying with a large average elevator angle up and all of a sudden a quantity of ballast is released, evenly distributed about the center of buoyancy, the ship becomes bodily light or lighter without change in static trim. However, the new lightness induces an aerodynamic change in trim which requires more elevator yet. If this is not applied or not available the ship actually dips down of its own accord and may descend rather than ascend. This is contrary to the expected result of dropping ballast evenly. Such a behavior appears as a reversal of buoyancy control and as such is of similar effect as a stall but evidently this condition is not an aerodynamic stall. It is the result of lack of adequate

[2] Revue Générale De L'Aéronautique, Vol. 14.

elevator control. It is interesting to note that this situation has an influence upon the radius of the circle of recovery from a dive, and that ships of this kind will require more height to check a dive when lightened by release of ballast amidships than by omission to do so. This was noted by C. P. Burgess. Both he and P. Changeux draw attention to the fact that the proper remedy in a case like this is not to drop the ballast amidships but somewhat forward. It is remarkable that the total leverage of the ballast dropped need not be very large. It is the natural decision, *viz.*, to drop ballast forward when one wants to raise the bow. In a ship of the *Akron* or *Macon* type the fins are so effective that the aerodynamic moment is relatively small and confined to moderate angles of attack. Therefore here the forward leverage of ballast desirable should be quite small.

In an airplane, quite aside from problems of controllability, the equilibrium of lift and weight, thrust and drag is not maintained independent of attitude or flight path slope. The reason is that, contrary to the buoyancy of gas, the lift of a wing tilts with the wing. As the airplane is zoomed into a flight path elevation angle ϵ, the lift leans over backward, and aside from reducing the vertical load bearing component at the rate of the cosine, it throws a sine-component aft on top of the drag, thus requiring additional power. This power is none other than that required by any hoisting device to hoist a given weight W at a given rate of ascent ($v \sin \epsilon$). It has to be furnished by the motor and propeller in excess of that power required to maintain level flight. (In the airship in equilibrium there was no weight to hoist.) Therefore an airplane can be zoomed into a stall.

There are four major ways to stall an airplane: (1) Incorrect trim (center of gravity too far aft); (2) Excess drag (control surfaces, spoilers, towing other aircraft, or lack of power, for instance trying to maintain height with motors in trouble); (3) Excess load (inertia forces, ice); (4) Lift and weight are orientated at detrimental angles to each other (zoom, steep climb, sharp curve). A combination of all four is for instance the steeply banked curve where the "lift" is slant, providing a horizontal component to counteract centrifugal force and leaving the vertical cosine component only to carry the weight. Temporary or inertia forces accompany vertical accelerations as for instance in the bottom of a loop or in levelling off or zooming up from a dive. They act exactly as a temporary increase of weight (item 3) and may lead to a stall.

Theoretically an airship may stall in the same way and from the same causes when the airship is heavy, as then the lift for the part of the weight in excess of buoyancy is produced by the same mechanism as on an airplane. The main difference in behavior lies in the great length of an airship which enhances the importance of problems of longitudinal stability and controllability. An airship when light is capable of gliding and stalling upwards, all aerodynamical phenomena then being inverted. While the heavy stall is dangerous inasmuch as it may strand the ship before recovery can be effected, the light stall checks itself when the ship arrives at its pressure height and begins to valve gas, thus automatically restoring buoyancy equilibrium. It entails the hazard of overclimbing by inertia and becoming heavy by excessive loss of gas.

When an airship is called upon to fly at an angle of pitch (attack) and create dynamic lift, this lift distributes itself by no means similar to the buoyancy but in a peculiar way. It consists of four parts: a good portion is borne by the forebody of the ship. The near cylindrical midship region gets almost nothing. The rear part of the ship's hull forward of the fins experiences a strange small share in the "wrong" direction, *viz.*, from leewards to windwards. The third part of the lift is borne by the fins and the part of the ship's stern between fins. The elevators carry the rest. The fin lift is the major part. The part carried by the elevator depends a great deal on the aerodynamic (and static) stability of the ship and on trim. With over-empennaged blimps the elevators are usually required up for up pitch so they experience only small forces in attack. Large rigid airships are usually so empennaged that the elevator is down for up pitch so that it carries a good part of the dynamic lift when the ship was in good trim. As a wing the ship's hull has both a very "poor" aspect ratio and very blunt tips, in fact nothing but two tips. The fins act more like stubby

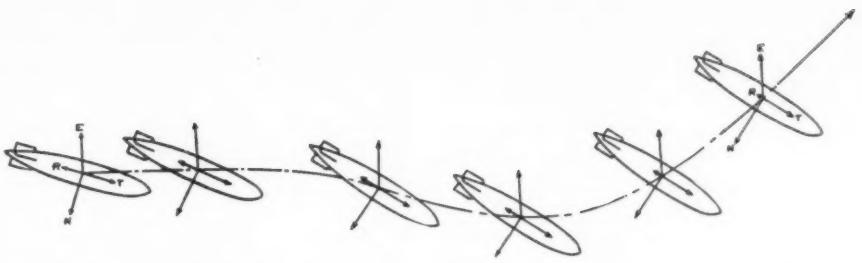

FIG. 3. Stall upward, light ship. *E*, excess buoyancy (positive); *N*, normal air force component; *R*, axial air force component; *T*, thrust.

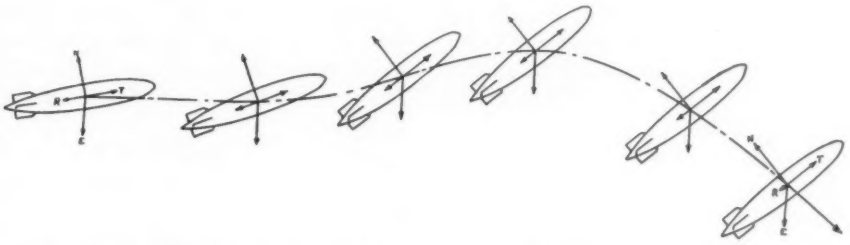

FIG. 4. Stall downward, heavy ship. *E*, excess buoyancy (negative); *N*, normal air force component; *R*, axial air force component; *T*, thrust.

airplane wings than the hull, and the tips are sharper but the aspect ratio is still mediocre. Evidently the loss of lift incurred with increasing angle of attack need not necessarily hit hull and fins at the same time.

Should the bow of a heavy ship's hull stall first, the ship would suddenly appear noseheavy; the nose would drop; the ship would now glide down exactly like an airplane[3] and pick up speed until this speed is sufficient to build up enough forward lift at an angle of attack below the stalling angle. Conventional airships do not behave like this; ships equipped with bow elevators or bow wings might.

When the stall begins at the empennage the ship may become longitudinally unstable and cannot be held under control. Ships without water recovery are known to have been so stalled when excessively light from fuel consumption. They would then be allowed to rise to pressure height and control would be regained even before buoyancy equilibrium was completely restored by automatic valving.

[3] Especially those of the Canard type.

However, this maneuver was perhaps not a stall in the sense of an actual drop of lift coefficient with increasing angle of attack but a case of limitation of controllability. When a light ship requires all of its available up elevator angle to fly at a negative angle of pitch (down by the nose) the least bit of further loss of weight will find it unable to balance the moment, the tail rises, the ship forces itself down by virtue of its own lightness. As a secondary result, as the ship dips, the rearward component of the surplus buoyancy slows down the ship and this additional drag is what finally leads to the stall upward (see Fig. 3). This is an everyday occurrence when a light ship makes an unsuccessful approach to land.

Similarly a heavy ship flying nose up, elevator down, may find itself unable to check a sudden rise and then stall downward, see Fig. 4. In both of these cases the loss of control could have been averted by trimming. Shifting ballast aft in the light ship and forward in the heavy ship would have restored plenty of elevator control range. Admittedly no captain will cherish the idea of risking finding his ship out of trim when

the engines should develop trouble. Nevertheless shifting of ballast in the cases just cited may be the preferable compromise if it may not only avert the loss of control but, should it even fail to do that, would at least greatly accelerate and facilitate recovery after the stall, thus averting immediate dangerous consequences.

A stall may not only be incurred unvoluntarily by lack of control range but it may be invited inadvertently by a precarious maneuver under full control. When properly trimmed and maintaining altitude at a given horsepower input the dynamic lift the ship can produce by maintaining an angle of pitch increases with this angle and with the square of speed. However, the drag increases, with small angles slowly, but with larger angles more rapidly. Therefore the speed the ship may attain under a given power drops with the angle of attack. Where this drop of speed begins to count more for the reduction of dynamic lift than the angle for its increase, the maximum lift is attained. Beyond this angle looms the stall. This type of stall is very flat or gradual. The lift carried at these angles of attack is tremendous. At full power (assuming an initial speed when in equilibrium of 74 knots) the *U S S Macon* is estimated to be capable of carrying dynamically 60,000 lbs.

With all aerodynamical characteristics known from wind tunnel tests one can readily compute the speed attainable under a given power for any angle of attack and then the lift carried at this speed and angle and then, by trial and error, determine the condition of maximum lift, aside, of course, from possible scale effect.

Analytically one can directly derive the maximum lift condition by the following approximation. Both the lift and the drag equilibrium are expressed each by an equation containing functions of two interdependent variables, *viz.*, velocity of flight and angle of attack; by eliminating one of the two variables, the lift can be expressed as a function of the other. The lift is a maximum when the derivative of this expression with regard to this variable is zero. The value of this variable (speed or angle of attack) for which this is the case, indicates the condition of the stall.

If the propeller thrust at open throttle would not vary with airspeed, the angle of attack of maximum lift can thus be readily found to be $\alpha_T^* = (c_0/n')^{\frac{1}{2}}$ in radians where c_0 is the minimum drag coefficient at zero angle of attack and n' the derivative of the (total) normal force coefficient (c_n) with respect to the angle of attack (α). (The two coefficients c_0 and c_n must be referred to the same dimensions.) For the Goodyear Zeppelin type of rigid airship, c_0 can be assumed to be about 0.02 (per $\text{Vol}^{\frac{1}{3}}$ and unit velocity head) estimated from power absorbed in speed trials, in accordance with wind tunnel results extrapolated for scale effect; n' can be taken from wind tunnel tests as 0.9 without elevator influence and 1.2 with such elevator influence as can be derived from previous full size flight experience as required for trim balance. Thus the stalling angle would be $9\frac{1}{2}°$ or $8\frac{1}{2}°$, respectively. The stalling airspeed is then $v_T^* = v_0(2^{-\frac{1}{2}})$ or 71 percent of the top speed (v_0) attained by the ship in buoyancy equilibrium under the same thrust.

In reality the propeller thrust increases as the speed is slowed down. If the power available were constant (which it is not) the thrust would vary inversely proportional to the airspeed. Under this assumption the angle of attack of the stall would be $\alpha_P^* = (3c_0/n')^{\frac{1}{2}}$ and the corresponding stalling speed $v_P^* = v_0(4^{-\frac{1}{2}})$ or 63 percent of top speed in buoyancy equilibrium under the same power.

In reality it is neither thrust nor power that stays unaffected by the variation of airspeed but torque, which again is connected with either of them via a varying propeller efficiency. Actually a variation of thrust results that is between the two fictitious extreme cases just studied. The variation is governed by the constancy of torque and the amount of slowing down of engine speed with slowing down of ship's speed. This actual propeller thrust-*vs.*-airspeed curve $T(v)$ can be very well approached by the most contiguous parabola

$$T = (t_s - tv^2)(\rho\ \text{Vol}^{\frac{1}{3}}/2)$$

where the first term is (very nearly the same as) the static thrust. With this notation the stalling angle is $\alpha^* = [(c_0+t)/n']^{\frac{1}{2}}$ i.e., the term t expressing the drop of thrust with increase of velocity head acts just as an increase in drag area. For the Goodyear Zeppelin type of rigid airship, t is of the order of 0.009 so that the stalling angle

becomes $10\frac{1}{4}°$ or 9°, respectively. The stalling speed is again $v^* = v_0(2^{-\frac{1}{2}})$.

Now it must be remembered that at this angle stalling takes place only when it is angle of attack and attitude at the same time. With an angle of inclination (attitude) much greater but the ship climbing it need not stall.

On the other hand, even with an angle of attack equal to the critical, stall ensues only if an attempt is made to maintain altitude or, what is worse, to climb when heavy or descend when light. The attainment of the critical angle merely signifies the ship can no longer maintain altitude with the power available and in this sense it is forced down (or up).

However, quite similar to an overloaded or underpowered airplane, it can now resort to a power glide and derive enough useful energy from the gradual loss of altitude to escape the uncontrolled stall. Of course the glide is only a temporary relief. With the light ship the upward glide is automatically terminated at pressure height without trouble; with the heavy ship, ballast will have to be jettisoned to regain sustained flight before the ship is stranded, or motor power increased.

When the ship is not so controlled as to try to maintain altitude, then the variation of the angle between the dynamic lift force and the surplus weight comes into play. The stalling angle of attack remains almost unchanged by a variation of flight path slope but at this angle of attack less speed may be attained and thus the maximum lift that can be produced is less. The approximate law governing this loss of lift in terms of the lift at zero climb (L) is

$$\Delta L/L = -n'\alpha\epsilon/[n'\alpha(\alpha+\epsilon)+c_0+i],$$

where ϵ is the angle of flight path slope against horizon, and $\alpha+\epsilon=\tau$ the angle of inclination (attitude).

If the heavy ship is made to climb or a light ship to descend, ϵ and α have the same sign and the danger of stalling is enhanced in proportion. If a heavy ship is allowed to lose altitude, although with the nose up, or if a light ship is allowed to gain altitude, although with the nose down, the danger of stalling is much deferred, ϵ and α then have opposite sign, and speed and maximum lift are increased.

If τ is zero, i.e., the ship held on even keel, ($\alpha=-\epsilon$) in which case when heavy it will sink, when light rise, there is no stalling in the true sense of the word. Such a condition is temporarily possible as long as there is altitude to sacrifice; this is a steady glide.

Should the slope of descent become very steep, then there may occur a secondary type of stall due to the reduced efficiency of the propellers which now work under pronouncedly slant inflow. This is expressed in a reduction of our quantity t_0 and has again the influence of reducing the speed and lift attainable at the stalling angle of attack.

Certain aircraft are built to be capable of being "mushed" or "squatted" down under extremely large angles of attack up to almost 90° where they descend more after the manner of a parachute. In this condition they acquire their vertical terminal velocity for which the resultant air forces carry the weight. Whereas heavier than air machines can, as a rule, be stalled into this manner of flight only by an accurately executed maneuver, a whipstall for instance, an airship can be deliberately brought into this condition by merely stopping it in midair. When in buoyancy equilibrium, the airship after having spent its momentum in coasting will hover on the spot, and, when in trim it will stay on an even keel. If now an appreciable quantity of ballast is dropped, or gas is valved, then buoyancy equilibrium is "ruptured" and the ship starts to descend or rise, as the case may be and finally attains terminal vertical velocity. The vertical motion is checked either by reestablishing buoyancy equilibrium, as for instance in landing through an inversion, or by resuming forward speed and creating dynamic lift. In the vertical descent the ship acts much like a parachute. The elevators are practically useless. Inclination can only be controlled by trimming with shifting loads. If in good trim before rupture of equilibrium, the bow will descend or rise faster than the stern, because the fins develop more parachute action than in proportion to the buoyancy share of the rear body of the ship. Therefore the vertical attack will automatically develop an axial reaction component and the ship will, of its own accord, pick up some forward speed. This rapidly inclines the resultant line of attack and may soon bring the ship under some degree of

elevator control. As more speed is picked up, the motion gradually turns into a regular glide. The reverse gradual change from glide into parachute descent may occur when a ship is initially heavy (or light) and then stopped to coast. The real vertical motion can thus be achieved only if the ship is heavy (or light) by the tail, otherwise forward speed will not completely die down.

If with any forward speed a heavy ship is allowed to point its nose down, or a light ship its nose up so that α and ϵ are of opposite sign, there is no stalling but the ship begins to pick up speed, just like a gliding airplane would. It so to speak sails on its surplus weight or buoyancy. (This boost of speed may become of importance for the recovery maneuver.)

A temporary apparent, not real, increase of weight can be produced by the inertia force evoked by a vertical acceleration such as centrifugal force in a zoom from a dive or from resistance to a vertical gust. A ship otherwise in buoyancy equilibrium cannot stall from this cause, it would merely be slowed down but continue the vertical curve. A heavy ship might theoretically be thus thrown into a stall, but practically it would be much more likely to stall only in a later phase of the maneuver, namely, when well on the climb.

The stall due to additional parasite drag is readily appraised by an increase of the drag coefficient. If c_p be the coefficient of additional drag, the stalling angle is actually increased to

$$\alpha^* = [(c_0 + c_p + t)/n']^{\frac{1}{2}},$$

but the total lift to be carried is reduced because the speed is reduced. In fact, it drops inversely proportional to the increase of the angle of attack.

Just how much is the maximum dynamic lift reduced by any particular increase in drag? This decrease is approximately proportional to the additional drag coefficient (c_p), *viz.*,

$$\Delta L/L = [(c_0 + t)/(c_0 + c_p + t)]^{\frac{1}{2}} - 1$$

or approximately $\Delta L/L \sim -\frac{1}{2}c_p/(c_0 + t)$ for relatively small parasite drag.

Such additional drag may become critical for instance when a ship in a near-stalling condition is suddenly called upon to turn.

In general a heavy ship in danger of stalling must be flown somewhat like an airplane in a similar condition: avoid taking the nose up and avoid turns or any loss or waste of power. The utility of a good dynamic lift indicator and especially an angle of attack indicator is well apparent.

There are thus two things to watch: (a) for a definitely heavy or light ship the angle of attack varies with speed, air density and trim; (b) the speed made good depends on power available, on flight path slope and on additional parasite drag. Thus the danger of stalling is approached when the heaviness (or lightness) approaches the maximum lift the ship can possibly carry and this maximum lift is not a constant but varies with maneuvers. The actual danger can always be indicated by measuring the actual angle of attack as it varies and by staying safely clear of the stalling angle of attack, provided the latter is well known. It again depends upon parasite drag, propeller characteristics and trim.

A ship may be heavy or light yet safely far from stalling. Nevertheless flight conditions may change rapidly and the stalling angle may sooner be approached than anticipated. For instance, the ship may run into air of different temperature and, the gas not following quickly in temperature, superheat is rapidly changed and buoyancy equilibrium destroyed. Or, the speed may slow down when engine trouble develops, for instance. On the other hand, a ship which stalled while flying at reduced speed may regain control upon speeding up the engines. As dynamic lift depends on air density a given heaviness or lightness and speed may be safe in one altitude but lead to a stall in higher altitude. A heavy ship stalling down may regain control as it falls into denser atmosphere whereas a light ship becomes the harder to check the higher it runs away. This density influence is so pronounced because the engine power is increased by greater air density practically proportional to the ship's drag and lift, while the top speed attainable is independent of air density.

A form of pseudo-stall is encountered when a ship climbs or lands through a strong vertical wind gradient or wind shift strata. When flying into the upper wind it descends into a lower tail wind or lesser head wind, relative air speed is

suddenly diminished and any dynamic lift will suddenly decrease and, if near the critical angle, may terminate in a stall.

An airship not equipped with stall-indicating instruments can find itself in a condition that may to the uninitiated look like, and have all the earmarks of, a stall, but is not a stall in the aerodynamic sense of the word and requires altogether different actions and remedies to cope with the possible imminent danger of the situation. It is the situation of being trapped in a vertically moving shaft of air in which the airship is bodily carried up or down according to whether it is a rising or falling current. Such vertical motion of the atmosphere is well known to assume powerful proportions where unstable stratifications or thermal gradients have built up from meteorological sources of energy. Meteorological altitude research and especially the recent exploits of soaring flights over great cross country distances and those accomplished after cutting loose from aircraft-tow ascents made for the special purpose of such investigations have revealed that vertical velocities of up to 3 m/sec. are quite frequently encountered in average flying weather and that vertical velocities from 3–6 m/sec. can be found in and underneath cumulus clouds and that appreciably higher vertically velocities are present in the dynamically active regions of thunderstorm areas and cold front formations.

After the airship has completely entered into such a "shaft" of air, assuming the latter to be much larger in extension than the ship is long, then the aerodynamical functions of whatever dynamic lift is created and of whatever pitching moments are to be derived from elevators, exclusively depend on the relative motion of the ship against this air and are in no way affected by the vertical motion of this air. Therefore a ship originally in buoyancy continues to fly at zero angle of attack, provided the temperature in the shaft was the same as the air temperature was when the ship was originally in buoyancy. If kept on an even keel it will bodily participate in the upward or downward motion of the shaft and thus approach its pressure height or the ground. There is no lightness or heaviness to be deduced from the observation that the altimeter shows rising or falling and there is no pressure difference created between the top and bottom part of the hull by the vertical current. The ship is merely in it and part of it and does not "feel" it. If the ship was originally light and flying with angle of attack, it can continue to do so without any change in attitude or equilibrium, and it will again be carried up or down with the same rate of ascent or descent as the current in which it flies. In practice there probably will be a concurrent change of air temperature because the descending shaft of air is probably colder than the surrounding air else it would not descend. Therefore the ship would derive a sudden superheat from the lag of its own gas temperature and thus become slightly light. In the course of the descent the lightness would become enhanced because of adiabatic compression of the gas which is subject to a higher expansion exponent than the outside air. Similarly a ship carried up would first lose superheat by entry into warmer air and then again by adiabatic expansion and become heavier. This phenomenon acts beneficially towards checking the unvoluntary change of altitude but the controllability may be adversely affected if the ship is in a regime where down elevator is required to trim a heavy ship and up elevator to trim a light ship.

On the other hand, if a ship is flying too low for comfort or too near pressure height unprepared for a sudden large discharge of gas, then it becomes essential to maintain altitude. With a ship in or near buoyancy equilibrium there should be ample resource in simply inclining the ship against the current. The simple law of geometrical (parallelogram) addition of velocities teaches that the vertical air velocity that can be checked is the product of the ship's air speed times the sine of the ship's flight path slope, which with zero angle of attack is the same as inclination. Thus with a ship's airspeed of say 30 m/sec. (58 knots) it takes all of 30° inclination to outmaneuver a 15 m/sec. vertical current. Even a current of 10 m/sec. would require an angle of pitch of $19\frac{1}{2}°$ under these circumstances. One is naturally reluctant to put the ship into such an unusual attitude, yet that is the proper thing then to do. Yet there need be no fear of stalling from excessive inclination. But recognition of the cause for the rapid descent (or

ascent) to be a vertical current and not to be suddenly acquired heaviness or lightness is of prime importance for the decision especially since there are only seconds available to reach a decision. If the ship's descent was caused by excessive loss of buoyancy then the maneuver of pitching it upward would be altogether wrong and could immediately lead into a stall.

If a ship finds itself in a vertical current while it is moderately heavy (or light), then the struggle against change of altitude is limited by the same angle of inclination (attitude τ) as would have led to a stall without the presence of a vertical current.

Thus far we have only considered what happens to a ship when it is bodily inside of a current of uniform vertical velocity. It remains to see what happens at the borders of such currents or gusts. Entry into a vertical current is not conducive to a stall towards the danger side. For instance, a heavy ship entering a down gust zone is dipped down thus making it pick up speed and relieve the angle of attack. A (very) heavy ship entering an up gust may have its nose thrown up enough to stall it but even so it would be bodily carried up.

A critical zone is the exit from the vertical current. Here a ship out of buoyancy equilibrium may be thrown towards a stall and one already stalled may either "recover" or else it may stay stalled and now suffer the unchecked consequences.

All of this brings home how essential it is that instruments are provided that do show immediately whether and to what degree any vertical motion is due to extraneous or internal causes, as the choice of the steps to be undertaken to its combat depends on this knowledge.

The theory and mechanisms entering into the design of such instruments are perfectly well understood. It is therefore merely a matter of devoting the necessary time to their construction and introduction in order to banish the uncontrolled stall of an airship into the realm of theory and to guard airship navigation from any possible hazard connected with inadvertent occurrence of this phenomenon. This development is already well on its way.

Metallurgy in Aviation

ALFRED DE FOREST, *Bridgeport, Connecticut*

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THE rôle which metallurgy has played in aviation, as in all other forms of transportation, is enormous. Instead, however, of a mere recital of progress made by all the forces of development, it may be of more interest to examine the mechanism by which this progress is made available to the aeronautic industry.

In the first place, what leads to improvement in materials, their treatment and application? Strangely enough, the requirements of engineers, impossible to fulfill, have been a source of much progress. The development of alloy steels in the automobile industry was forced by demands from engineers wholly free from the restraints of the conventions of metallurgy. The steel companies, seeing tonnage and therefore profits in sight, were forced to study metallurgy and continuously to improve the working properties of their steels.

The early automotive engineers were mostly far too busy to employ research metallurgists. When American steel makers protested too vigorously against exorbitant demands, the engineers went to Europe, particularly to Krupp, where fundamental metallurgy had long been used to advance the steel maker's art. Occasionally the industry supplied its own demands, as in the case of Wills and molybdenum steels.

Insistent demand slowly led to research in the steel producing industry, and the center of metallurgical gravity located in the schools under Howe and Sauveur from 1900 to 1915, has shifted toward the steel industry; first to the small specialized concerns such as Taylor-Wharton, Halcomb, United Alloy Steel, and lately to the U. S. Steel Corporation itself. The automotive industry as consumers have forced the steel producers into fundamental study, but it was done by way of a third party, the purveyors of the steel making alloys. Before the steel producers were paying much attention to alloy steel, the International Nickel Company was devoting real research to nickel steel, the United Carbon and Carbide Company was studying chrome, vanadium and molybdenum were building sales arguments and all were bombarding the steel buyers on the one hand and the producers on the other.

Aviation has greatly profited by the metallurgical improvements made necessary by automotive needs. It is now making new demands in its own right and the important question to consider is how these may most rapidly be carried out. Two distinct rôles are filled by the consumer's engineering-metallurgical skill. The first is routine metallurgical control of present available materials and processes. This requires a plant metallurgist to write and uphold specifications and care for the every-day metallurgical needs of the company. Second is the advisory capacity to obtain as closely as possible, the ideal demands of the engineer. Here the metallurgist must know what can be done with present materials when pushed to their utmost and where to look for further improvements when one line of progress is blocked.

In many cases at the present time three types of material and construction are in competition; first, heat-treated alloy steels; second, light alloys of aluminum or magnesium; third, high tensile cold-worked stainless steels put together by spot-welding. Experts in each of these fields can advise on the special applications of the different methods of construction, but a high degree of skill and experience is necessary in choosing the best possible type of design and material for any particular purpose. Frequently experiments must be run to determine the relative possibilities and, at this point, the metallurgist's advice becomes important. A few experiments based on fundamental principles may cost far less than a large number of hap-

hazard tests under widely varying and uncontrolled conditions. For example, a choice between materials can often be decided upon from simple tests in a specially designed fatigue machine. This method is usually cheaper than an airplane engine and plane as a proving-ground for novelties. It requires, however, careful analysis of working conditions, to be sure that the experimental answer is really based on the same factors which obtain in practice.

It is unfortunate that standardization is so readily applied and so easily accepted. As soon as standardized laboratory tests are set up, the general tendency is to accept at face value these results and to write specifications which in time become a hindrance and not a help. Standard test factors come into government specifications and progress is halted by arbitrary requirements.

The aviation industry differs particularly from all others in that the various agencies of the Federal Government are exceedingly active in engineering development. The N.A.C.A., the Bureau of Standards, the experimental departments of the Air Corps and the Bureau of Aeronautics, contribute very largely to engineering progress. There is therefore a tendency to leave to these agencies development work which in other cases is carried on by industry itself. This state of affairs can easily become a disadvantage rather than an aid to progress.

In ship-building, the demands of the Insurance Companies limit the naval architect to certain types and weight of construction in the commercial field. In naval work on the contrary, experimental methods can be tried out and great progress has resulted from the application of new methods and materials, as in the pocket battleship of the Germans with welded construction, and in the American Navy where spot-welded stainless steel and aluminum have been applied in a totally unconventional manner. Another suggestive example is the new light construction of railway equipment. The Master Car-Builders' specifications required $\frac{1}{2}''$ diameter solid steel handles on doors. But in the light weight trains where all the old specifications are waived, such handles can be of aluminum tubing of one-tenth the previous weight. If the Car-Builders' specifications had not become so rigid, it is possible that progress towards reduction in weight would have begun long ago.

The aeronautical engineer is privileged to ask for what he wants and he may be expected to know his requirements more thoroughly than the producer. He will be best served if he has at his disposal technical knowledge not inferior to the producer, with whom he is dealing. At the present time, the latter is generally in advance, both in research personnel and facilities. This lead however was established before aviation became important, and it is reasonably certain that research personnel trained in the government bureaus and the forward looking technical schools, will at some future time find employment in the aeronautic industry.

The Behavior of Wings in Yaw

SERGIO DEL PROPOSTO, *Aachen*

(Received January 1, 1934)

INTRODUCTION BY L. HOPF, AACHEN

THE position of an airplane relative to its path is not entirely determined by the angle of incidence when the path is not in the plane of symmetry of the airplane. An additional angle is required to fix the asymmetry. The air force, therefore, depends upon the angle of yaw in addition to the angle of attack. In the range of normal flight conditions this dependence is so small that it can be neglected in practical considerations. However, for large angles, at or above the angle of maximum lift, appreciable rolling and yawing moments result from the cross-wind. These moments play a large part in the motions at high angles of attack and are decisive in starting and developing the spinning motions. Also, these moments are of the same order of magnitude as the moments caused by the rotation of the wing. The study of these moments, started only recently, has not been completed. Unfortunately, there are no systematic measurements on airfoils available over a sufficiently wide range of angles of attack. Results of this kind are necessary in order to arrange computations and establish a theory. Furthermore, the question of the influence of the plan form of the wing and the shape of the wing tips on the moments had to be answered. I suggested, therefore, to Mr. del Proposto a systematic test of these phenomena. Dr. Wieselsberger kindly placed the equipment of the Aerodynamic Institute at our disposal for these tests.

The most important results of this work are as follows:

(1) Numerical values for rolling and yawing moments of an airfoil on which to base the calculation of the spinning motion and the dimensions of the vertical tail surfaces.

(2) Proof that the moments arise because of two distinct physical effects which can be experimentally separated.

(3) The first, associated with the wing tip corners (Fig. 1), is explained as the result of unusual profiles subjected to the wind. This effect can be strongly influenced by shaping the wing tips.

(4) The second effect is larger than the first and appears even on an airfoil without corners. This effect can be understood qualitatively from a consideration of the boundary layer. No attempt was made to influence this effect. The question of whether the rolling moment should be kept small in practice cannot be answered unconditionally for in certain cases this moment might tend to prevent spinning.[1] There will be cases, however, for which the same moment might accentuate a spinning motion. It is, however, advisable in every case to reduce the unstablizing yawing moment.

* * * * * *

THE force and moment coefficients of an airfoil are, when the influence of Reynolds number is neglected, functions of the position of the airfoil relative to the wind as determined by the angle of attack and the angle of yaw. If the angle of yaw is different from zero the resulting force will generally not act in the plane of symmetry of the airfoil and it then becomes necessary to determine all three of its components as well as the three moments. Under such conditions an airfoil is said to be yawed or side-slipped.

The influence of side-slip with special attention devoted to the effect on the incipient spin has been discussed.[2] Data on full scale airplanes and on airfoil models tested at small angles of attack are presented elsewhere.[3] In the present paper the author extends the airfoil data to larger angles of attack, studies the effect of wing tips, and attempts to give a physical explanation of the phenomena resulting from the side-slip.

[1] L. Hopf and S. del Proposto, Abh. a. d. Aerodyn. Inst. d. T. H. Aachen, No. 10.

[2] L. Hopf and S. del Proposto, Zeits. f. Mechanik No. 13, p. 393; Abh. a. d. Aerodyn. Inst. d. T. H. Aachen, No. 10.

[3] R & M Nos. 831, 932, 933 and 1059.

Influence of the Wing Tips

Calculations by Blenk, based on Prandtl's airfoil theory show, for an airfoil in yaw, a rolling moment proportional to the angle of attack and acting in such a direction as to cause the upstream wing tip to move downward. Unsymmetrical distribution of the induced downwash velocity along the span causes a decreased lift on the upstream wing tip as compared with the downstream wing tip. Blenk's results agree with experiment only for small angles of attack while for the most important range of larger angles of attack they disagree even as to the direction of rotation.

It was felt that this disagreement might possibly be caused by the neglecting of the influence of the wing tip corner in Blenk's calculations. See Fig. 1. The author therefore undertook to measure the distribution of forces along the span, especially near the wing tips, on a small rectangular airfoil with a Göttingen No. 420 profile. The procedure was to measure the pressure distribution on a number of cross sections for different angles of attack β and angles of yaw $\gamma=0$, ± 20 ($+20$ for the wing tip upstream and -20 for the wing tip downstream). See, for example, Figs. 2 and 3. It was possible to obtain accurate curves showing the distribution

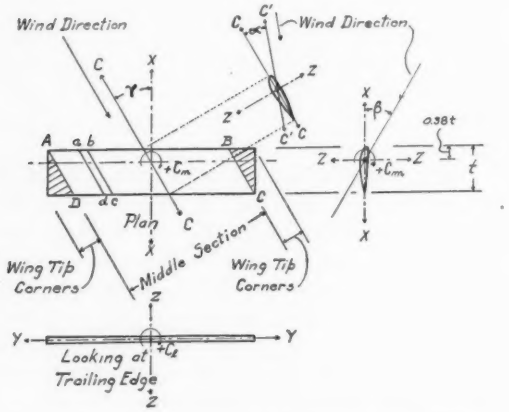

Fig. 1. The angle of attack (α) is the angle between the velocity vector and its projection on the x–y plane. The angle of attack (β) is the angle between the projection of the velocity vector on the x–z plane and the x–x axis. The angle of yaw (γ) is the angle between the velocity vector and its projection on the x–z plane. The angle of yaw (τ) is the angle between the projection of the velocity vector on the x–y plane and the x–x axis. (Not shown.)

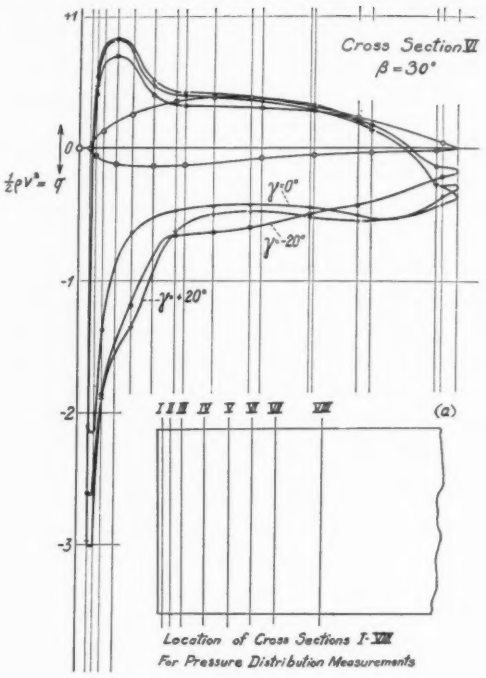

Fig. 2.

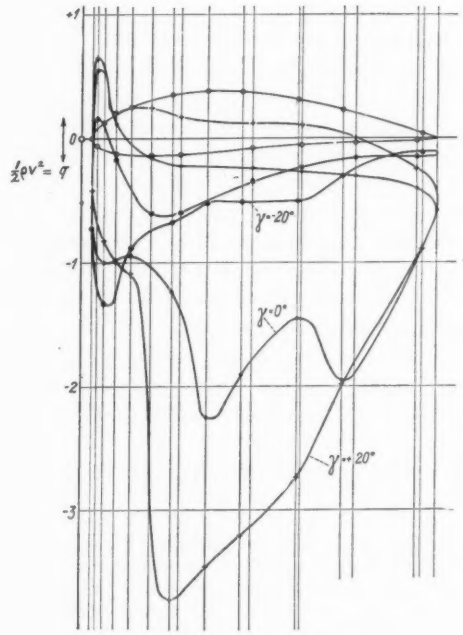

Fig. 3. Cross section I. $\beta=30°$.

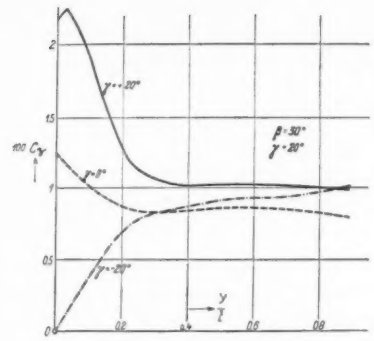

FIG. 4. C_s distribution at the wing tips, plan form No. 1.

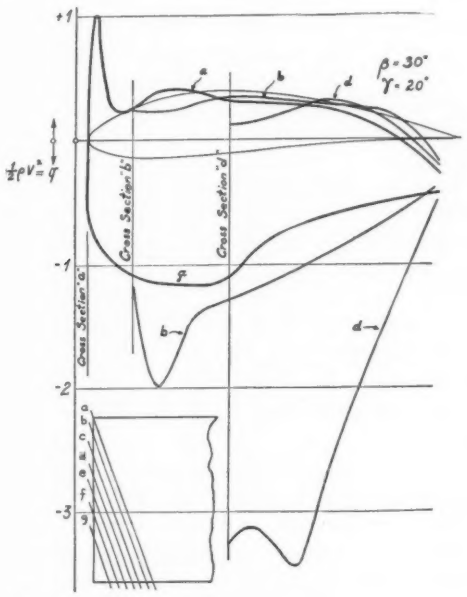

FIG. 5. Sketch for the interpolation of the pressure distribution on cross sections *a*, *b*, etc., corresponding to a cross-wind angle of 20°.

of the normal force along the y–y axis near the wing tips. From the curve shown in Fig. 4 it can be seen that the wing tips contribute toward offsetting the rolling moment as calculated by Blenk. Because of the large number of cross-sections in which pressure distributions were measured it was possible to interpolate and determine the pressure distribution in sections parallel to the wind direction and thus to demonstrate clearly that the large negative pressures are due to the sharp leading edge of the

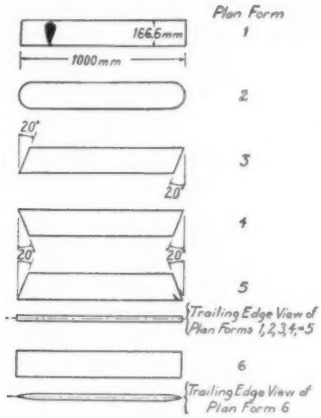

FIG. 6. Plan forms. Profile Göttingen No. 420.

wing profiles at the wing tips subjected to the wind. See Fig. 5. It follows then that the shaping of the wing tips will influence the behavior of a wing when there exists a cross-wind.

Although this "wing tip corner" effect furnishes a moment in the correct direction its magnitude is not large enough completely to explain the discrepancy between Blenk's calculations and experiment. In order to determine quantitatively the magnitude of the discrepancy a further investigation of wings in yaw was undertaken in which the actual forces and moments were measured. One series of tests was made with wing tip corners, Fig. 1, cut off so as to duplicate as nearly as possible the conditions assumed by Blenk for his calculations. A second series was run with varying wing tips as shown in Fig. 6 at angles of attack up to 50 degrees and at angles of yaw up to 25 degrees.

ROLLING MOMENT

Fig. 7 shows a comparison between the calculated rolling moment (dotted line *a*) and that actually measured (C_l) for a wing with plan form (3) shown in Fig. 6. The moments are taken about the x–x axis and are considered positive when the upstream wing tip tends to lift. Referring to Fig. 7 it can be clearly seen that only for small lifts do the theoretical calculations give fairly good agreement. In order to arrive at the causes for the deviations at larger lifts it is convenient to analyze the rolling moment by

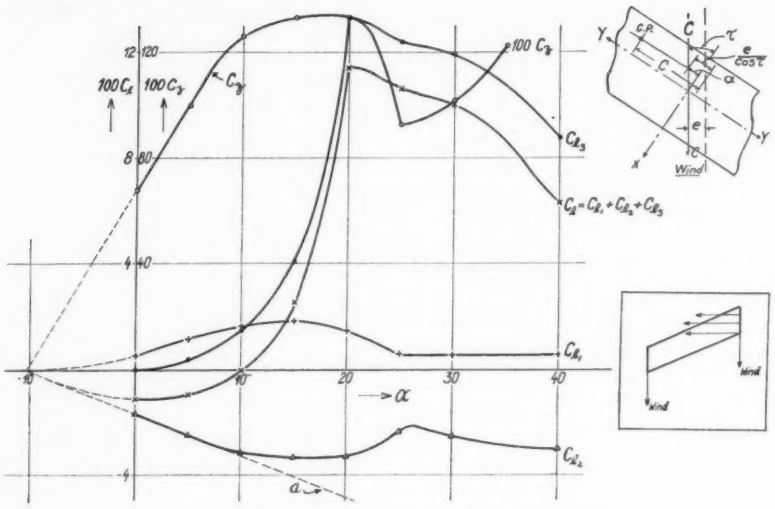

FIG. 7. Rolling moments of wing No. 3.

dividing it into three parts. The first part assumes symmetry of the downwash velocity along the span in which case the resulting normal force would act in the plane of symmetry through axis c–c and produce a moment around the reference axis x–x given by

$$\frac{M_{l_1}}{(s/2)v^2sb} = C_{l_1} = \frac{a}{b}\frac{Z}{(s/2)v^2s} = \frac{a}{b}C_z,$$

where a is a function α; Z force component in the Z–Z direction; s the wing area; b the wing span.

The second part considers the unsymmetrical distribution of the downwash velocity along the span which shifts the center of pressure to a line parallel to c–c but displaced from it.

$$C_{l_2} = -C_z(e/\cos\tau)(1/b).$$

In order to bring the resultant center of pressure to the location actually measured it is necessary to add a third term which can be written in the form

$$C_{l_3} = (C_z/b)C,$$

where the distance C is determined from experiment as a function of the angle of attack.

The total C_l will then be

$$C_l = (C_z/b)(a - e/\cos\tau + C).$$

Fig. 7 shows the three contributing moments, C_{l_1}, C_{l_2}, and C_{l_3} as functions of the angle of attack. The third part C_{l_3} begins to have an important effect at those angles of attack where the coefficient of normal force C_z deviates from the theoretical straight line. This seems to indicate that the difference between the theoretical and the actual moments can be explained by the separation of the flow and that this separation of the flow occurs earlier on the downstream part than on the upstream part of the wing.

The following consideration of the boundary layer of an airfoil in a cross-wind leads to a probable explanation for the asymmetry of the breakaway. Because of the loss of energy, particles near the surface cannot proceed far against a large increase in pressure. There results then an accumulation of retarded particles which displaces the flow from the surface and brings about a separation.

The airfoil theory differentiates between the two-dimensional and the three-dimensional theory by taking into account for the latter case a changing angle of attack along the span. The influence of a flow normal to the wind direction caused by a small negative pressure gradient from the wing tips toward the center is entirely neglected. This negative pressure gradient from wing element to wing element is quite large in the case of a wing subjected to a cross-wind

because of the staggering of the elements. See Fig. 7. The streamlines of the potential flow outside the boundary layer are only slightly inclined because here the velocity is high. In the case of the particles in the boundary layer the condition is such that the particles have lost their energy and will, therefore, follow the pressure gradient. The result is then a flow of particles in the boundary layer from the upstream section of the wing toward the downstream part and an increased boundary layer thickness in this region. The tendency for a separation on the downstream section of the wing is, therefore, highly increased. The results as shown in Fig. 7 indicate that the rolling moment due to the asymmetric separation is very much larger than the one calculated by Blenk. As the separation spreads and reaches both wing tips with increase of angle of attack, the asymmetry and rolling moment become smaller again. A few of the results obtained on airfoils with varying wing tips are shown in Figs. 8, 9 and 10. The rolling moment C_l is given as a function of the angle of attack β for $\gamma = 5°$, 15° and 20°. A comparison between airfoils No. 1 and No. 6 is of special interest, for in No. 1 the wing tips were sharp edged while in No.6 the wing had tips which were tapered in profile. In further comparison of the two airfoils it will be noticed that the rolling moment is generally smaller for airfoil No. 6 than for airfoil No. 1 but that the maximum rolling moment is higher in the case of No. 6. The explanation for this lies probably in the separation of the flow at the sharp edged wing tips at low angles of attack in the case of wing No. 1, while for wing No. 6 the flow is able to follow the contours when the angle of attack is much larger.

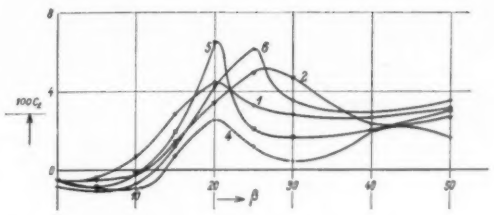

FIG. 8. Comparison of the rolling moment of wings of different plan forms. $\gamma = 5°$.

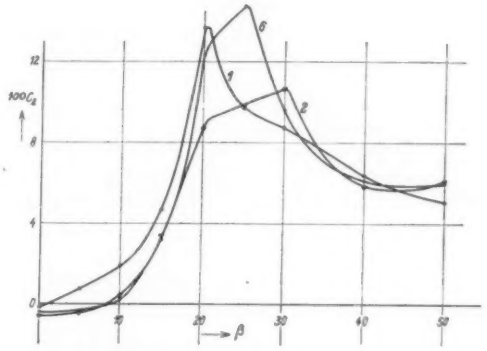

FIG. 9. Comparison of the rolling moment of wings of different plan forms. $\gamma = 15°$.

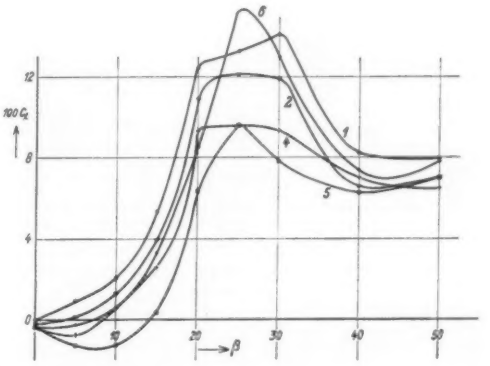

FIG. 10. Comparison of the rolling moment of wings of different plan forms. $\gamma = 20°$.

YAWING MOMENT

The yawing moment is caused chiefly by the tangential force and partly by the side force. The tangential force acts in a direction opposite to the wind as long as the flow does not separate and increases with increasing lift for any airfoil element under consideration. Since the lift distribution is asymmetric for an airfoil in yaw and since it is larger on the downstream part of the wing than on the upstream section at fairly small angles of attack, the tangential force will be larger on this part and will try to turn the wing into the wind. With changing lift distribution and a shifting of the larger part of the lift to the upstream section of the wing the yawing moment will become negative and continue to increase in a negative direction because the tangential force also will change its sign with a breaking away of

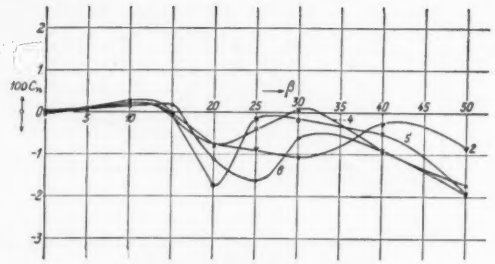

FIG. 11. Comparison of yawing moment for wings of different plan forms. $\gamma = 5°$.

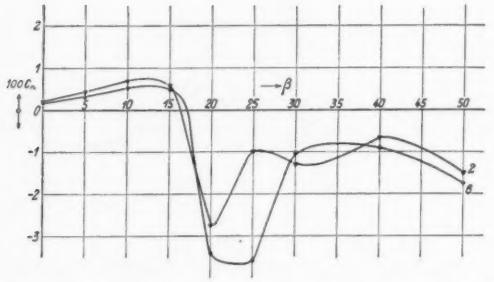

FIG. 12. Comparison of yawing moments for wings of different plan forms. $\gamma = 15°$.

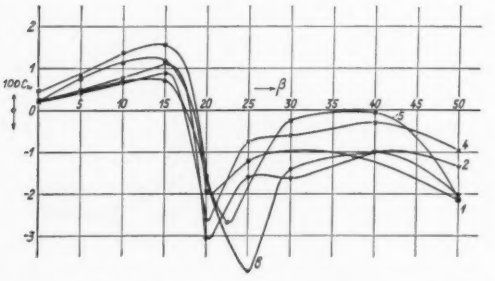

FIG. 13. Comparison of yawing moments for wings of different plan forms. $\gamma = 20°$.

the flow which occurs on the downstream part of the wing first. Figs. 11, 12 and 13 show the yawing moment plotted against angle of attack for $\gamma = 5°$, 15° and 20° for airfoils with varying wing tips.

PITCHING MOMENT

The pitching moment is only slightly influenced since the cross-wind scarcely changes the force distribution along the chord but only along the span.

SUMMARY

Systematic tests were made to study the behavior of an airfoil subjected to a cross-wind for angles of attack up to 50° and angles of yaw up to 25°.

Pressure distribution at a number of cross sections near the wing tips was measured to study the influence of the wing corners. The rolling moment due to the wing corners is too small to explain the discrepancy between measured moments and those obtained by theoretical considerations.

A further study on an airfoil without corners suggested that the largest effect, at least in the neighborhood of the stall, is due to the fact that the separation at the downstream part of the wing occurs before that at the upstream part, the resulting lack of symmetry of the lift along the span naturally leading to a strong rolling moment.

A number of airfoils with different plan forms were tested and compared to show the influence of the corners and the mutual interference with the middle section.

The knowledge obtained in the explanation of the rolling moment can be readily applied in analyzing the causes of the measured yawing moments. The fact that the yawing moment at low angles of attack tends to decrease the angle of yaw and at large angles of attack to increase the angle of yaw has been explained in this way. The tendency to increase is dangerous since it decreases the directional stability of the airplane.

The Oil Cooling Problem in Aircraft Engines

HAROLD CAMINEZ, *Allison Engineering Company*

(Received May 17, 1934)

The increase in the specific power output of aircraft engines has generally resulted in the need of lubricating oil radiators in aircraft engine installations. This paper discusses the effect of oil temperatures on engine performance and reliability. The temperature rise in the lubricating oil which determines the requirement for oil radiators is shown to be affected by the engine operating conditions. Test results are described which show how the heat dissipated to the lubricating oil is affected by engine speed, horsepower output, oil inlet temperature, cylinder jacket temperature and specific fuel consumption. The results of these tests indicate that engine speed and oil inlet temperature have the greatest effect on oil temperature rise and therefore determine the requirements for the oil radiator.

THE demand for greater power plant reliability has necessitated greater care in the design of the oil system in aircraft engine installations and has required means for maintaining the oil temperature within prescribed limits. The increase in the specific power output that has resulted with engine development has generally caused a higher oil temperature rise in the engine and has made the use of oil radiators more prevalent in modern aircraft engine installations. The data for determining the requirements of the oil radiator are not generally available. The data presented in this paper which were obtained from engine tests and bear on the oil cooling problem, should therefore be of general interest.

The lubricating oil should be maintained within certain temperature limits to safeguard the operation of the engine. When the oil temperature is too low, as in starting a cold engine, the engine friction is increased and mechanical efficiency of the engine is reduced as shown in the curves of Fig. 1. As seen from these curves, the change in friction for a reasonably wide temperature range is not great so that the usual variations in oil inlet temperature obtained in operation have but little effect on the engine power output.

The oil flow through the engine bearings is also affected by the oil temperature as shown in Fig. 2. The oil thrown out by the crankshaft bearings decreases as the oil temperature is reduced with the result that at too low a temperature there is danger that enough oil will not be thrown out to lubricate the parts which depend upon splash for their lubrication. This danger is present during the warming up period so that the throttle should not be opened wide until the oil temperature is above a certain specified minimum.

The characteristics of the lubricating oil vary with temperature as shown in Fig. 3. From this figure it is seen that there is a marked decrease in viscosity with increase in oil temperature. In aircraft engines the bearings are designed and lubricated so that the loads are supported by an

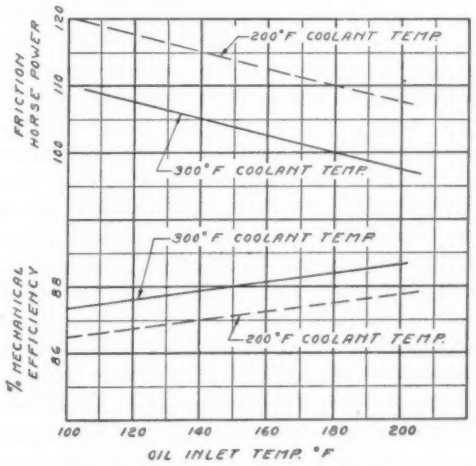

FIG. 1. Friction horsepower and mechanical efficiency *vs.* oil inlet temperature at 2400 r.p.m.

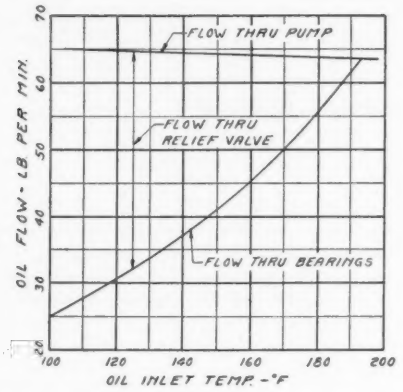

FIG. 2. Oil flow *vs.* oil inlet temperature at 2400 r.p.m. Oil pressure—130 lb. per sq. in.

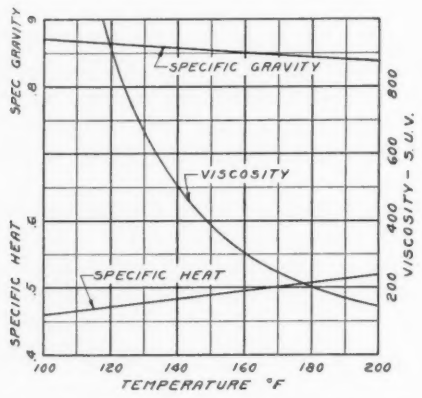

FIG. 3. Viscosity, specific heat and specific gravity *vs.* temperature. Gulf Pride No. 120 oil.

oil film under normal operating conditions. This oil film separates the journal from the bearing, prevents undue friction heat being generated at the bearings, and by separating these parts prevents wear from occurring during normal operation. The load that a bearing will support at a given engine speed without the breaking down of this oil film varies as the viscosity of the oil in the bearing so that as the viscosity decreases the load carrying capacity of the bearing is reduced. The minimum allowable viscosity of the lubricating oil in the bearing depends upon the unit load, the surface velocity, the bearing fits, and the design of the parts. Since viscosity is a function of temperature with a given lubricant, it is seen that there is a certain maximum oil temperature which must not be exceeded in any given engine in order to have safe bearing operation. This safe maximum temperature will vary with different engines depending largely on the unit loading and the detailed design of the bearings.

Bearing failures also occur in engines due to the crushing of the bearing metal. The crushing strength of the bearing metals decreases with increased temperature and where the unit bearing loadings are high this factor must be considered. With Babbitt metal this decrease in strength with temperature is great in the temperature range usually employed in aircraft engines so that such bearings cannot generally be employed in aircraft engines when high oil temperatures are used. The bearings lined with high lead bronzes are not affected to the same extent by temperature and appear to have adequate crushing strength to support all the normal loads occurring in aircraft engine bearings throughout the range of oil temperatures now employed.

Another factor that limits the maximum safe oil temperature is the fact that the lubricating oils break down more readily at higher temperature. This breaking down of the lubricating oil is one of the causes of piston ring seizure. Some lubricating oils break down more rapidly at high temperatures than others so that when high oil inlet temperatures are employed, care should be used in selecting a lubricating oil that will not carbon up too much and cause seizure of the piston rings. In some aircraft engines the maximum allowable oil inlet temperature is limited more by the tendency towards piston ring seizure than by bearing failure.

In a normal installation the lubricating oil undergoes a temperature rise when flowing through the engine. This heat absorbed by the oil is dissipated in the oil system of the airplane. The capacity of a given installation to dissipate heat increases with the rise in oil temperature so that the oil temperature is stabilized at the point where the heat dissipated in the airplane oil system is equal to the heat absorbed by the oil when flowing through the engine. In order to design the airplane oil system, it is therefore necessary to know the oil temperature that must not be exceeded and also the quantity of heat

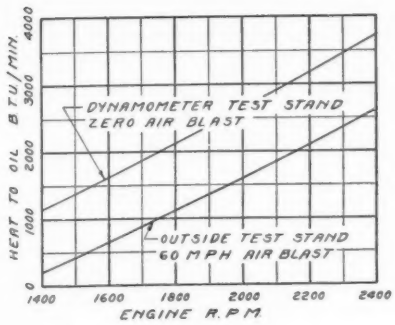

FIG. 4. Heat to oil *vs.* engine r.p.m. on propeller load.

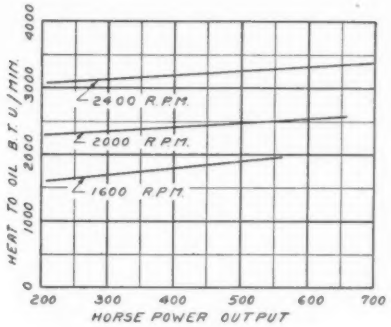

FIG. 5. Heat to oil at various r.p.m. *vs.* horsepower output.

that must be dissipated. The inlet oil temperature to the engine is the temperature that is generally measured and regulated. The maximum inlet oil temperature for safe operation is determined in test by the engine manufacturer. Modern engines using Grade 120 oil (viscosity 120 S U V at 210°F) when operating in hot climates usually specify a maximum inlet oil temperature of 185°F (85°C). For winter operation a Grade 98 oil is used in which case the specified maximum inlet oil temperature is 167°F (75°C).

The heat to be dissipated by the lubricating oil is influenced to a great extent by the engine installation. The oil in the engine picks up heat from the bearings and the cylinders and gives up heat to the crankcase and oil sump, so the temperature of these parts influences the quantity of heat which the airplane oil system must dissipate. Fig. 4 shows the quantity of heat to be dissipated by the oil system for two installations. These data were obtained from tests of a 12 cylinder high temperature liquid cooled engine. The upper curve shows the heat to the oil in dynamometer tests under propeller load where there was no air blast on the engine. The lower curve was taken from tests on an outside test stand where a 60 mile per hour air blast was maintained on the engine and the other conditions of test were identical with that of the dynamometer test. The dynamometer test duplicates installation conditions where the engine is completely enclosed by an unventilated cowling, while the outside test stand duplicates conditions where there are sufficient louvres in the cowling to allow an air blast of 60 miles per hour on the engine. It is seen from these curves that with the engine used in this test the oil radiator required for the outside test stand installation need only be 70 percent as large as that required in dynamometer tests.

Dynamometer tests were run on this twelve cylinder liquid cooled engine at various fixed speeds and with varying power outputs. In these tests the rate of oil flow and oil temperature rise were determined and the heat given to the oil was computed. From the data obtained in these tests, curves in Fig. 5 were drawn which show the heat to the oil at various power outputs and engine speeds. From these curves it is seen that the quantity of heat that the oil absorbs in the engine depends mainly upon engine speed and is only slightly influenced by engine power output. Therefore, engines that operate at high crankshaft speeds generally require larger oil radiators than engines of the same power output which operate at lower crankshaft speeds.

The effect of coolant jacket temperature and inlet oil temperature on heat absorbed by the lubricating oil was studied in this same engine dynamometer installation. The engine was operated at constant speed and constant power output and the heat absorbed by the oil was determined for various oil inlet temperatures and coolant outlet temperatures. The results of these are shown graphically in Fig. 6. From this figure, it is seen that the heat absorbed by the lubricating oil increases as the coolant jacket temperature is increased and that the heat to the oil is reduced as the oil inlet temperature

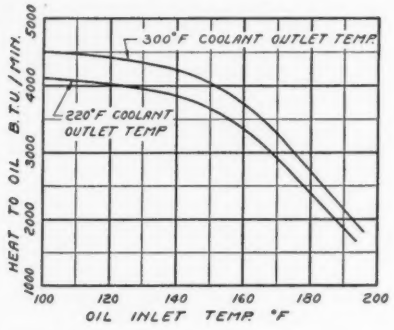

FIG. 6. Heat to oil *vs.* oil inlet temperature.

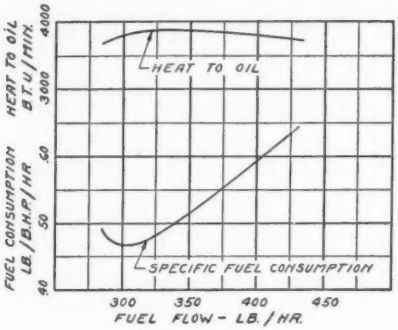

FIG. 7. Heat to oil and specific fuel consumption *vs.* fuel flow.

increases. The oil inlet temperature has considerably more effect on the heat absorbed by the oil than the coolant jacket temperature. Referring to Fig. 6, it is seen that the heat to the oil with 160°F oil inlet temperature and 220°F coolant outlet temperature is the same as that with 168°F oil inlet temperature and 300°F coolant outlet temperature.

The effect of specific fuel consumption on the heat absorbed by the lubricating oil was also studied. In the dynamometer tests with fixed speed and throttle position, the mixture control on the carburetor was adjusted from the full rich to the full lean position and the heat dissipated to the oil was determined. The results of these tests are shown in Fig. 7 which shows that leaning out the mixture to the point of maximum economy increases slightly the quantity of heat absorbed by the lubricating oil. In the test described, the quantity of heat was increased about 3 percent when adjusting the carburetor from the full rich position to the position of maximum fuel economy.

The data cited in this paper show that the heat to be dissipated by the lubricating oil increases rapidly as the crankshaft speed is raised and decreases as the oil inlet temperature is increased. An analysis of the tests cited shows that the oil temperature rise in the engine appears to be caused mainly by the frictional heat generated in the engine. In order to reduce the size of oil radiator or to eliminate the necessity for same, engines should be developed to produce their rated power outputs with lower crankshaft speeds or they should be designed to operate reliably with higher lubricating oil inlet temperatures. Developments in lubricating oils which will allow them to operate at higher temperatures with less carbonizing and with less tendency to cause piston ring seizure are also desirable.

Present Views on the Nature of Detonation in the Otto Cycle Engine*

C. F. TAYLOR AND E. S. TAYLOR, *Massachusetts Institute of Technology*
(Received January 31, 1934)

BY "detonation" is meant the phenomenon in the Otto cycle engine which causes the "pinking," "spark knock" or "carbon knock" familiar to all automobilists. Recent laboratory researches have shown that this characteristic sound is always accompanied by the following phenomena:

1. An extraordinarily rapid rate of pressure-rise in the last part of the charge to burn.

2. High frequency pressure waves within the cylinder gases, starting near the end of the combustion process.[1]

The present most satisfactory physical theory is that detonation is simple compression-ignition, or "auto-ignition" of the last portion of the charge to burn. This would easily account for the apparently infinite flame-speed indicated by photographs such as those of Withrow and Boyd,[2] since the compression-ignition process usually forms ignition points at many places in the charge, almost simultaneously. Ricardo[3] first proposed this simple theory, but until recently, it was not generally accepted because of its failure to explain why all fuels do not knock in the inverse order of their compression-ignition temperatures. This theory has recently been re-affirmed by the work referred to above, and E. S. Taylor[4] has offered the following explanation, which shows that no real contradiction exists.

It has been demonstrated by Tizard and Pye[5] that a mixture of fuel and air has no definite ignition temperature, but that it has an ignition-temperature-time relation. That is, if a combustible gaseous mixture is heated by sudden compression to a given temperature, *after a certain definite time* it will ignite. If the temperature to which it is compressed is increased, the time before ignition will be reduced. Thus for a given combustible mixture a curve of temperature against time-lag or "delay" may be drawn as in Fig. 1. Different fuel-air mixtures have

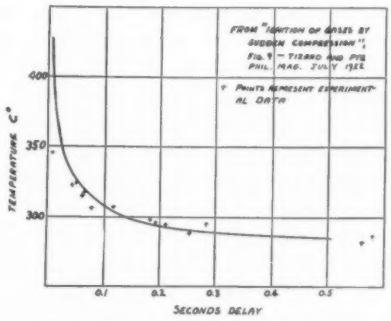

FIG. 1. Theoretical curve of ignition delay *vs.* compression temperature for heptane and air.

different temperature time characteristics, but all exhibit a decreasing "delay" with increasing temperature.

Detonation in an engine becomes more severe as the speed is reduced, even though the volumetric efficiency is kept constant, and the spark advance is set for the best power (with non-detonating fuel) in each case. Marvin and Best[6] have shown that under these conditions the rate of flame travel is nearly proportional to the r.p.m., and the only considerable change in the

* Presented at the Annual Meeting of the I. A. S.

[1] C. F. Taylor, et al, *A New Instrument for the Study of Combustion*, J.S.A.E. **34**, 59, February (1934).

[2] Withrow and Boyd, *Photographic Flame Studies in the Gasoline Engine*, Ind. Eng. Chem. **23**, 539, May (1931).

[3] Ricardo, *The Internal Combustion Engine*, Blackie and Son, Vol. II, 1923.

[4] E. S. Taylor, *The Importance of Auto-Ignition Lag in Knocking*, N.A.C.A. Tech. Note No. 452, March, 1933.

[5] Tizard and Pye, Phil. Mag. [6] **44**, 79, July (1922) and [7] **1**, 1094, May (1926).

[6] Marvin and Best, *Flame Movement and Pressure Development in an Engine Cylinder*, N.A.C.A. Tech. Note No. 399, 1931.

cycle is in the time of burning. It is well known that an engine equipped with two diametrically opposite spark plugs will detonate more when one plug is not operating, even though the maximum pressure and temperature occur later and are therefore lower. The effect of turbulence in accelerating the flame travel and suppressing detonation is also common knowledge. These facts lead to the inescapable conclusion that, *in the detonation process there must be a time effect.* In further confirmation of this, we may recall the fact that certain fuels with low ignition temperatures will run without preignition in a high-speed engine. Tizard and Pye[5] give the ignition temperature of normal heptane as 298°C. Assuming a compression exponent of 1.35 and a compression ratio of 3.75, a simple calculation in which due allowance is made for the products of combustion of the previous cycle will reveal the rather astonishing fact that the temperature of compression may exceed the ignition temperature found by Tizard and Pye for normal heptane. It is known, however, that normal heptane can be used in an engine with this compression ratio without causing knock. Granting the presence of even a very short time lag, the explanation of this fact is very simple. Although the temperature of the last part of the charge to burn is raised to the point of self-ignition at the end of the compression stroke, *and considerably above this point before it finally is consumed*, it does not have sufficient time to complete the necessary physical or chemical preliminaries to self-ignition before the flame front passes through it and it is completely burned.

We now have a complete picture of what happens to this "last part of the charge to burn." It is first compressed by the action of the piston, and then by the combined action of the piston and the burning portion of the mixture, further compressed to a point which may be considerably above its minimum "self-ignition" temperature. Whether or not it knocks, depends upon the temperature-time-lag characteristics of the fuel. If it is kept at a sufficiently high temperature for a sufficient length of time, it will knock. If not, the flame front will pass through it and it will participate in the normal combustion.

It will be noted that the significant temperature in determining whether or not a given mixture detonates, is the highest temperature reached by the last part of the charge before its combustion. If this is above its minimum "self-ignition" temperature, and if the normal flame does not pass through it too soon, it will detonate. While all fuels show a decreasing time-lag with increasing temperature, the shape of the temperature-time-lag curve may vary considerably. Thus, the ignition temperatures determined in an experimental bomb may show fuel *a* to have a higher self-ignition temperature than fuel *b*, while under the different temperature and time conditions in an engine, the reverse may be the case, and fuel *a* will detonate more readily. This apparently explains the contradiction between the "self-ignition" temperatures and knocking tendencies of certain fuels.

The combustion of the last part of the charge during knock is extremely rapid. Because of the inertia of the gas itself, there is insufficient time for the volume to increase appreciably during combustion, and under these conditions burning approaches the constant volume condition. This results in a high local pressure in the last part of the charge to burn, and excites the resonant frequencies in the gas within the cylinder which have already been mentioned.

So much for the physical aspects of detonation. The chemical aspect, i.e., the details of the chemical process by which compression ignition occurs, together with the chemical explanation of the action of knock-suppressors and knock-inducers, is not within the province of these remarks. There is still considerable difference of opinion among the experts regarding chemical details, but when agreement is finally reached, I believe it will be found that knock inducers and knock suppressors operate on the ignition-temperature time-lag relation of the mixture.

Metal Fatigue and Methods for Its Measurement*

R. CAZAUD, *Service des Fabrications de l'Aeronautique, Paris*

(Received February 28, 1934)

THE study of the fatigue of metals used in aeronautical construction is of considerable interest on account of the fluctuating loads to which aircraft parts are subjected. The French Air Ministry has, accordingly, sponsored a broad investigation of the nature of fatigue failures and a determination of the causes of rupture, the influence of fatigue resistance on the suitability of metals for particular purposes, the fatigue limits of different metals, the influence of holes and incisions on the fatigue limits, and the possibility of including fatigue measurements in the routine testing of metals. A resumé of the studies undertaken and the practical results obtained is presented in this report.

I. Examples of Fatigue Failures in Airplane Engine Parts

In the area of fracture of a piece that has failed through fatigue, a smoothly polished outside zone may be distinguished from a more or less veined and granular inner one. The general absence of any appreciable elongation in the vicinity of the break suggests a failure of great suddenness.

An examination of the faces of the fracture usually reveals cracks starting from defects inside of the section, or from irregularities on the outside surface, and suggests that the failure originated in a minute crack which gradually widened under repeated stresses until there no longer remained enough sound metal to support the applied loads. The smoothness of the outer zone is due to a prolonged wearing of the edges of the break on each other, and the granular inner zone is produced by the final rupture. Sometimes it is possible to distinguish successive smooth zones when the failure is due to several intermittent exposures to fluctuating loads.

The illustrations show typical fatigue failures of pieces either broken in service or by a rotating flexural testing machine. The similarity between the fractures of the test pieces and the breaks that occurred in actual use is to be noted. In some cases the smooth zone covers a large part of the section; in others it is very narrow. The starting of cracks on the borders of a hole and their gradual advance inwards is clearly shown in Figs. 1 and 2. The failure of the crank shown in Fig. 3 is of a similar nature; it is due to a fatigue crack starting from both sides of a lubricating hole. Another crankshaft failure is illustrated in Fig. 4; in this case, the development of a fatigue crack, which started at the root of the thread,

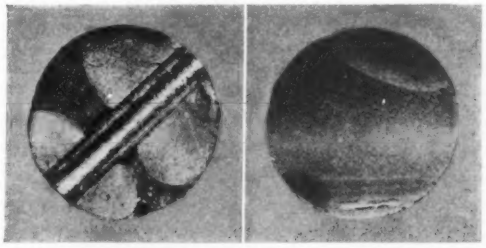

FIG. 1. FIG. 2.

FIGS. 1 AND 2. Fractures of fatigue test pieces.

FIG. 3. Crank failure.

* Presented by General Sabatier, translated by Professor Manfred Rauscher.

FIG. 4. Crankshaft fracture. (Starting point of failure indicated by *a*.)

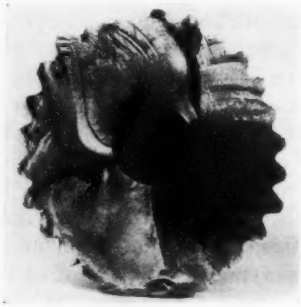

FIG. 5 Spline failure.

FIG. 6. Failure of a light alloy propeller.

was aided by the sharpness of the groove. The spline of another crankshaft is shown in Fig. 5. Finally, Fig. 6 shows a light alloy propeller that failed through fatigue at the junction of the blade and the hub; insufficient filleting was the main cause of this failure.

These examples indicate the detrimental effect of holes and sharp angles. They suggest the importance of testing the fatigue strength of pieces with incisions. They also show the necessity of a full understanding by the designer of the effects of holes and incisions on the strength of parts that are to be subjected to variable loads.

II. The Nature of Metal Fatigue

Without going into the details of the various theories of fatigue, it may be mentioned that fatigue is definitely known to be a process of progressive cracking. The existence of cracks in parts broken by fatigue, as well as in pieces that have been subjected to a large number of repeated stresses without actual breakage, has been thoroughly established by microscopic metallography. It has been found that the cracks, starting in the neighborhood of physical defects, especially of non-metalic inclusions, generally follow lines of least resistance in the metal, often running from one inclusion to another. In iron, cracks tend to run across ferrite crystals and only rarely affect the junctions of these crystals. On the other hand, in certain alloys like duralumin it is found that the cracking is intercrystalline and the failure is brought about by the gradual separation of the crystals (Fig. 7).

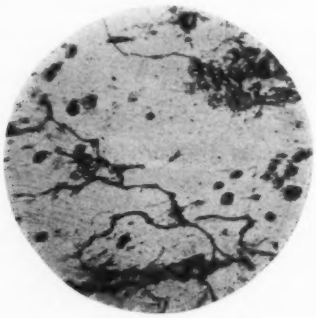

FIG. 7. Fatigue cracks in dural.

The investigation of the cracks in their initial stage is a very delicate matter. The study is facilitated if it is possible to confine cracking to definite cross sections by the use of special threaded test pieces. The cracking then begins at the bottom of the thread and its progress may readily be followed. The microphotograph of Fig. 7 was obtained in this manner.

III. Determination of the Fatigue Limits of Metals

Parts subjected to variable loads are commonly designed only for static load conditions, safety factors being used to allow for possible fatigue

effects. These factors are sometimes too low, causing failures in operation, and sometimes too high, resulting in excessively heavy and bulky construction. The rational approach is clearly to test the materials under working conditions or under conditions as similar as possible to them.

An especially simple way of producing alternating tensions and compressions is by the method of rotating flexure:

Principle of the rotating flexure test

The test piece is held at one end in a rotating clamp; the other end of the piece carries a weight suspended on a ball bearing. On account of the bending, each fiber of the piece is thus subjected to a stress that follows the sinusoidal law with a mean value equal to zero. At a given section the greatest stress variations naturally occur in the outside fibers, which consequently suffer most from fatigue. Between different sections, the stresses generally differ; but by giving the test piece a conical shape, it is possible to secure almost uniform maximum stresses over the middle portion of the piece, and to restrict failures to a narrow region about the middle (Fig. 8).

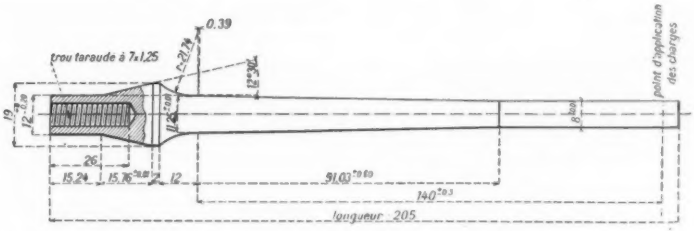

FIG. 8. Standard rotating flexure test piece. (Dimensions in mm.)

Test procedure

The fatigue limit may be defined as the maximum amplitude of alternating stress which the material is able to sustain indefinitely. Rigorously speaking, this limit cannot be determined, and it is considered sufficient for practical purposes to run the tests over 30–100 million stress cycles, a range large enough to cover most actual cases.

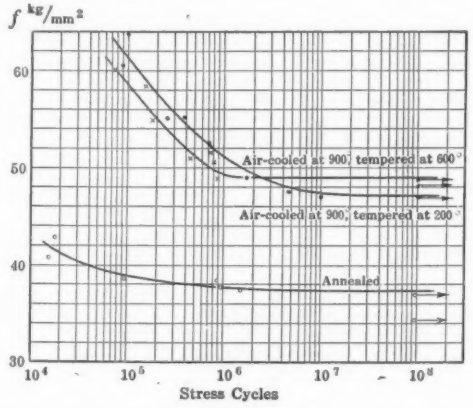

FIG. 9. Fatigue curves for a self-hardening chrome-nickel steel.

The practical fatigue limit is then determined by making experiments on a series of test pieces of the same material with different attached weights, and noting the number of stress cycles which lead to failure in each case, the weight in any given test being kept constant. It may be observed in passing that tests should be run with decreasing loads rather than with increasing ones, since the fatigue limit appears more quickly in the former case and may be determined with fewer test pieces. The results are conveniently represented by plotting stresses on an ordinary scale against the number of stress cycles on a logarithmic one. The asymptote of each curve then gives the practical fatigue limit for the particular material (Fig. 9).

Results

Tests made by the French Aeronautical Research Service since 1929 have covered all the metals currently used or capable of use in aeronautical construction. Table I contains some of the results for various steels. These data allow the designer to select the steel and the heat treatment most suitable for a given member or

TABLE I. *Fatigue limits and general characteristics of steels.*

Steel	French standard designation	Chemical Composition								Heat Treatment	Mechanical Characteristics						Micrographic Structure	Fatigue Limit f	$\frac{f}{\sigma_{ult.}}$	$\frac{f}{\sigma_{yield}}$
		C	Mn	Si	S	P	Ni	Cr	Mo		σ_{yield}	$\sigma_{ult.}$	Elong. %	Red. of area	Imp. resis.	Hardness Δ				
Extra soft	11	0.10	0.45	0.03	0.08	0.03				Annealed at 950°; cooled in air	28	41	3.1		30.2		*Ferrite* and pearlite	26	0.63	0.92
Soft	12	0.19	0.42	0.26	0.006	0.04				Annealed at 900°; cooled in air	26	45.2	3.0		12.8	118	*Ferrite* and pearlite	23	0.54	0.88
Medium	13	0.29	0.65	0.25	0.046	0.06				Annealed at 875°; cooled in air	35	54	2.8		10	146	*Ferrite* and pearlite	26	0.48	0.74
Medium	14	0.36	0.25	0.16	0.015	0.012				Annealed at 850°; cooled in air	36	59	24.2	60	13.1		*Ferrite* and pearlite	27	0.46	0.75
Do.	14	0.36	0.25	0.16	0.015	0.012				Quenched in water at 850°; without tempering	150	170	7	48	3.5	450	Martensite	44	0.26	0.29
Hard	15	0.60	0.70	0.18	0.016	0.048				Annealed at 800°; cooled slowly	42.5	73.5	19	38	4.7	205	Ferrite and *pearlite*	30.5	0.415	0.72
Extra hard	16	0.65	0.64		0.025	0.037				Annealed at 800°; cooled slowly	45	80	17		21	230	Pearlite	32.7	0.41	0.71
Nickel	21	0.10	0.30	0.07	0.020	0.015	2.10	0.03		Quenched in water at 850°	41	54.5	17.5	78	24		Ferrite and pearlite	35.4	0.66	0.84
6% nickel	22	0.12	0.26	0.24	0.013	0.023	6.15	0.40	0.15	Quenched in water; untempered	125	144	9	58	11.3	360	Martensite	37	0.255	0.30
Do.	22	0.12	0.26	0.24	0.013	0.023	6.15	0.40	0.15	Quenched in water; tempered at 600°	116	137	9.5	49	10.4		Martensite and osmondite	44	0.32	0.38
23% nickel	23	0.65	0.85	0.10	0.016	0.04	22.95			Quenched in water at 950°	24	70	52	65	15	178	Austenite	31.2	0.445	1.30
Chrome nickel (medium)	31	0.19	0.50	0.23	0.010		3.00	0.95		Quenched in oil at 825°	120	152	11.4	41	10.2		Martensite	48.5	0.32	0.41
Chrome nickel (medium)	32	0.34	0.25	0.14	0.005	0.019	2.45	0.85		Quenched in oil at 850°; tempered at 200°	160	180	10	45	6.1	485	Martensite	46.5	0.26	0.29
Do.	32	0.34	0.25	0.14	0.005	0.019	2.45	0.85		Quenched in oil at 850°; tempered at 500°	108	114	11	62	12.4	365	Martensite and osmondite	43.5	0.38	0.40
Chrome nickel (hard)	33		0.35	0.27	0.018	0.02	3.00	0.50		Quenched in oil at 850°; tempered at 200°	120	180	7.5	46	5.2	540	Martensite	54.5	0.30	0.45
Do.	33		0.35	0.27	0.018	0.02	3.00	0.50		Quenched in oil at 850°; tempered at 650°	67.5	90	17	56	15.1	260	Osmondite	51.2	0.57	0.76
Chrome nickel (Self-hardening)	34	0.35	0.20	0.20	0.004	0.018	2.76	2.71	0.40	Quenched in air at 900°; tempered at 200°		183	8.5	50	7.1	470	Martensite	65	0.35	
Do.	34	0.35	0.20	0.20	0.004	0.018	2.76	2.71	0.40	Quenched in air at 900°; tempered at 600°	92	114	15.8	62	14		Martensite and osmondite	45	0.395	0.49
Chrome molybdenum		0.17	0.51	0.21	0.006	0.010		0.74	0.38	Quenched in water at 829°; tempered at 200°	120	144	5.7	52	9.2	418	Martensite	65	0.45	0.54
Do.		0.17	0.51	0.21	0.006	0.010		0.74	0.38	Quenched in water at 829°; tempered at 500°	100	109	10	68	12.5	310	Martensite and osmondite	52	0.485	0.515
Stainless with 13% chromium		0.25	0.45	0.67	0.021	0.017		13.8		Quenched in oil at 950°; tempered at 500°	130	160	9.5	33	2.1	435	Martensite	75	0.47	0.57
Stainless with 18% chromium and 8% nickel		0.17	0.52		0.008		8.25	17.15		Quenched in air at 1100°; untempered	23.5	66.5	56	76	33	130	Austenite	23	0.35	0.98

Note: σ_{yield}, $\sigma_{ult.}$, and f in kg/mm² (multiply by 1420 to get lb./in.²); impact resistance is Mesnager resilience in kg/cm² (multiply by 14.2 to get lb./in.²); hardness obtained by diamond point method.

TABLE II. *Fatigue limits of light metals.*

Metal	Condition	Mechanical characteristics σ_{yield}	Mechanical characteristics $\sigma_{ult.}$	Elong. %	Fatigue limit f	$\frac{f}{\sigma_{ult.}}$
Pure aluminum	Drawn	12	15.5	10	6.5	0.42
Aluminum-copper	8% Cu sand cast		12		5.5	0.45
	12% Cu die cast		15		4.8	0.32
Aluminum-silicon	5% Si die cast		15		4.6	0.30
	12% Si die cast		22		5.5	0.25
	18% Si die cast		20		7.1	0.35
Duralumin	Drawn and heat-treated	27.5	39.5	13.5	12.4	0.31
Magnesium alloy 6% Al	Drawn	17.2	27.6	18	11.4	0.41
Magnesium alloy 3% Cu 9% Al	Drawn	19.8	28	8.5	13	0.46

part. It will be noted that the highest fatigue limits are found among nickel-chrome-molybdenum steels, for which the fatigue limit is as high as 71 kg/mm². For 13 percent chromium steel the limit even reaches 75 kg/mm².

As for heat treatment, it appears that tempering after quenching is usually very beneficial, especially if the reheating temperature is around 600°C, corresponding to a homogeneous osmonditic structure. In numerous instances the fatigue limit of a tempered hard steel is higher than that of the same steel untempered, in spite of the considerable decrease in the elastic limit and static breaking strength brought about by the tempering.

Table II gives a few of the results obtained with light alloys. It appears that some of the magnesium alloys have fatigue limits as high as and even higher than the alloys with an aluminum base.

IV. Inspection Tests of Fatigue

These collected data encourage the consideration of the rotating flexure test as a standard routine test and the introduction of the fatigue limit into the French Aeronautical Stress Code. Two kinds of tests have been proposed:

First, an initial test for a new material consisting in the determination of the fatigue limit for 30 million stress cycles in the case of steels, and 100 million in the case of non-ferrous metals and alloys. This test is to be carried out with test pieces as shown in Fig. 8. If the material is not available in such a large size, the test is made on a smaller piece, as shown in Fig. 10, the dimensions being just one-half those of the standard test piece.

Secondly, an inspection test to verify the material's ability to stand up under one million stress alternations of definite amplitude. The dimensions of the inspection test piece (Fig. 11) permit its being cut out of almost any part used in aircraft construction.

V. Influence of Holes, Fillets and Incisions on the Fatigue Limits of Various Metals

(a) *The influence of holes* has been studied on standard test pieces with transverse holes of 1.15 mm diameter at a distance of 55 mm from the fillet, this being the region of maximum stress.

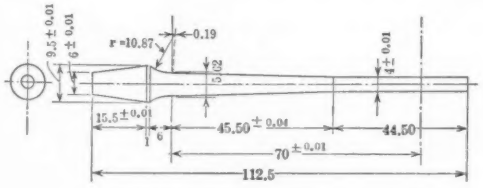

FIG. 10. Small type rotating flexure test piece.

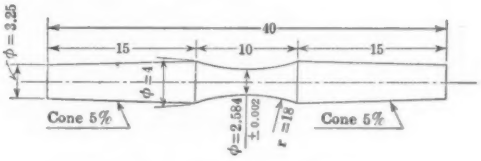

FIG. 11. Inspection test piece.

Table III gives some of the results of these tests, the stresses referring to the net section.

(b) Three *types of threading* have been tried: triangular with sharp bottom (Fig. 12a), triangular with rounded bottom (Fig. 12b), and half-round, corresponding to French railroad practice (Fig. 12c). In all cases the outline of the test piece at the bottom of the threads is that of the standard test piece. For a soft annealed steel with $\sigma_{yield}=29$, $\sigma_{ult.}=40$, $\Delta=120$: (Table IV). The triangular thread with the sharp bottom is seen to be slightly more harmful than the one with the round bottom or the half-round one, but the reduction in the fatigue limit is of the same order in all three cases. For heat-treated steels, the effect of each kind of threading depends greatly upon whether the threading was done before hardening or after. The pronounced influence of the internal stresses set up by the quenching is indicated by the following results obtained with pieces which had been threaded before hardening (see Tables V and VI).

The extremely low values for threaded pieces are due to the high internal stresses which the hardening produces at the bottom of the threads. They emphasize the danger of cutting the threads, or of making any kind of incision, before quenching. Since the hardness of the metal prohibits threading after quenching, a final reheating is absolutely necessary to relieve the internal stresses due to the quenching. For the

TABLE III.

Metal	Fatigue limit: Test piece without hole	Fatigue limit: Test piece with hole	Reduction in f %
Nickel-chrome steel, annealed: $\sigma_{yield}=41$; $\sigma_{ult.}=64.5$; $\Delta=175$	35	20	43
Nickel-chrome steel, quenched in oil and tempered at 650°: $\sigma_{yield}=67.5$; $\sigma_{ult.}=90$; $\Delta=260$	51.2	26	49
Nickel steel, quenched in water and tempered at 600°: $\sigma_{yield}=65$; $\sigma_{ult.}=81$; $\Delta=235$	38.8	25	35
Heat-treated duralumin: $\sigma_{yield}=30$; $\sigma_{ult.}=43.7$; $\Delta=114$	12.8	7.7	40

TABLE IV.

	Fatigue limit f	Reduction in f %
No threads	23	—
Sharp triangular threads	17	26
Rounded triangular threads	18	22
Half-round threads	18.5	20

TABLE V. *Steel with 0.2 percent C, 3 percent Ni, 0.8 percent Cr, quenched in oil at 825°, untempered;* $\sigma_{yield}=110$, $\sigma_{ult.}=130$, $\Delta=380$:

	Fatigue limit f	Reduction in f %
No threads	46.5	—
Sharp triangular threads	5	89
Rounded triangular threads	11	76
Half-round threads	16	65

TABLE VI. *Steel with 0.34 percent C, 2.45 percent Ni, 0.85 percent Cr, quenched in oil at 850°, tempered at 200°;* $\sigma_{yield}=160$, $\sigma_{ult.}=180$, $\Delta=485$:

	Fatigue limit f	Reduction in f %
No threads	46.5	—
Sharp triangular threads	11	76
Rounded triangular threads	17	63
Half-round threads	20	57

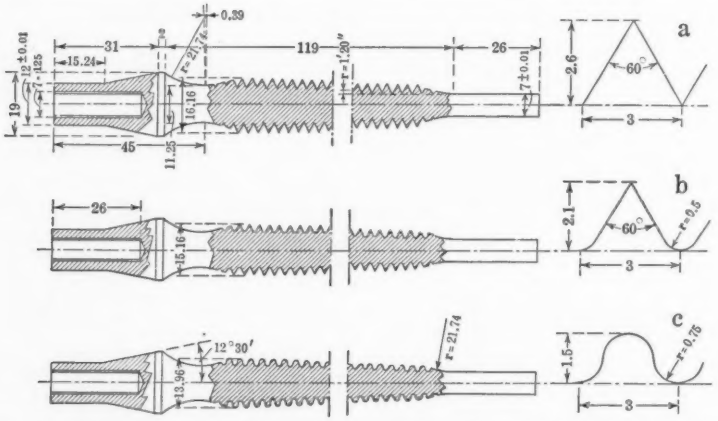

FIG. 12. (a) Test piece with sharp triangular threads. (b) Test piece with triangular threads rounded at the bottom. (c) Test piece with half-round threads.

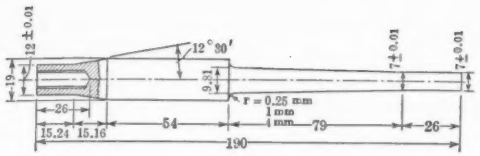

FIG. 13. Filleted rotating flexure test piece.

TABLE VII.

	Fatigue limit f	Reduction in f %
No threads	42	—
Sharp triangular threads	21	50
Rounded triangular threads	24	43
Half-round threads	26	38

second one of the steels mentioned above, tempering at 650° after quenching in oil at 850° gave $\sigma_{yield}=69$, $\sigma_{ult.}=83$, $\Delta=260$, with the fatigue limits as given in Table VII.

(c) The fatigue effects of *abrupt changes in section* are the same as those of incisions. The influence of the filleting depends on the ratio of the fillet radius to the smaller diameter. This influence has been studied by means of the test piece shown in Fig. 13, in which the section is reduced more or less suddenly by one-half, the fillet radii being 0.25, 1 and 4 mm. For a chrome-nickel steel with 0.19 percent C, 3.30 percent Ni, 0.92 percent Cr, quenched in oil at 825°, with $\sigma_{yield}=105$, $\sigma_{ult.}=133$, $\Delta=365$, the fatigue limits given in Table VIII were obtained.

These data relate to specimens machined after quenching. When the machining is done before quenching, the effect of the filleting is even more pronounced.

TABLE VIII.

	Fatigue limit f	Reduction in f %
Fillet radius 21.6 mm (Standard test piece)	46	—
Fillet radius 4 mm	44	4
Fillet radius 1 mm	33	28
Fillet radius 0.25 mm	27.5	40

VI. CONCLUSIONS

1. It appears from these tests that the rotating flexure test is eminently suited for the determination of the fatigue strength of metals.

2. An exact knowledge of the fatigue limits and of the reasons for the various form factors (for holes, incisions, etc.) is a prerequisite for the use of appropriate factors of safety.

3. The rotating flexure test appears bound to become, in the near future, one of the basic tests for materials used where close figuring is necessary.

COMMENTS OF A. V. DE FOREST

This paper treats many of the problems of repeated stress met with in aeronautics. Especially important is the warning that the designer should have a full understanding of the effect of holes and incisions on the strength of parts that are to be subjected to variable loads. The difficulty lies in the fact that little fundamental testing has been done on concentrated stresses.

A great deal of time has been spent on the determination of the endurance limit of different metals, particularly steels, using polished test specimens. The work of H. F. Moore at the University of Illinois is particularly well known in this field. Very little testing has been done on the effects on endurance of notches, fillets and holes, and yet in aeronautical materials such disturbances almost always locate the failures which occur in practice. It is not sufficient to know that the metal in question has a certain specified fatigue strength, it is also important that the proportion of this strength which is lost at a notch shall not be inordinate.

This timely paper suggests that an endurance test be made part of a material inspection specification for certain metals. Unfortunately at the present time, at least in the United States, there is not enough information as to the effect of notches on the different steels available. No standard forms have been evolved. The cold work of the metal surface at the bottom of the notch depends greatly on the method of preparation, whether cut by a tool or by a grinding wheel. As shown in this paper, internal stress may be difficult to deal with if the notch is cut before heat treatment. The tables of figures given do not tell how many specimens of each type were tested, nor the variation between the average and the extremes. Much evidence must accumulate to show that test variables other than those due to the metal itself have been reduced to insignificance before the test could fairly be used to reject metal otherwise acceptable.

The great difficulty in such a program lies in the question of approximating in test form the preparation, heat treatment, condition of surface and internal stress which prevail in the finished part. There are too many important variables in the finished engine part to ignore them by testing unrepresentative samples of metal. There is also no way, without much study, of applying under test conditions the exact type of loading to which a part is subjected in practice. It seems probable that conventionalized fatigue tests on full size parts could be worked out, but the design of the part rather than the material would undoubtedly turn out to be the deciding factor in its performance. After the best design had been established, it would be time to investigate the improvement possible by variations in the metal and its methods of preparation.

In spite of the objection that arbitrary fatigue tests of samples of material would not tell the whole story, the effort spent in such tests on notch specimens would greatly add to our knowledge.

Lessons from Naval Architecture

H. C. RICHARDSON, *U. S. Navy Department*

(Received April 26, 1934)

WHEN the subject of this article was proposed to the writer, it struck him as almost impertinent, considering the contributions that aerodynamics are effecting principally in the superstructures of modern seagoing vessels. But, after jotting down some of the things well known to the naval architect, he was reminded of certain features, some of which border on the realm of the Marine Engineer, to which the aeronautical engineer might well give greater consideration than is yet apparent in his product.

It is only natural that naval architecture makes its most intimate contact in hull design and performance with particular reference to seaplanes, but with no mean bearing on airship design, structural and aerodynamic.

The model basin has already contributed a big share in performance of hulls, and has made important contributions to forms of struts and balanced control surfaces.

The writer introduced the trochoidal wave method of hull stress analysis into naval specifications, which forms the basis of the Department of Commerce Specifications, and of the approximate alternative analyses.

The dynamic forces encountered by hydroplaning aircraft hulls differ so greatly from those experienced by immersed hulls of the displacement type, that the specified method is far from exact. There appears to be little data available other than tank tests and flight tests of the N.A.C.A., which by no means cover the full scope of the conditions met in service. But they have proved a basis of initial control which has admitted considerable advance.

It is impossible to visualize, much less specify, the probable forces to be encountered in taking-off, alighting on, or driving through a seaway; although some engineers have confidently set up and solved the effects of imaginary forces on their hull designs. In such a case, it is wise to set up something like a Lloyd's inspection, with a set of factors, to be modified by experience. Undoubtedly such in effect has been the result of the application of the "barn loading" and the trochoidal wave analysis of existing specifications.

The streamline theories have their foundation in hydrodynamics. In the airship these lessons have been heeded. For airplanes these teachings are only partially heeded. Fuselages are generally too attenuated; they might very well be of fuller form, with due influence being given to considerations as to directional and longitudinal stability characteristics. Improved structure with weight economies and better disposable load accommodation can probably be attained with, at the same time, improvement in performance. One is reminded of the excellent performance of the Breguets which won the Pulitzer Trophy, and of the early Voughts with their propeller axis on the axis of a well-formed fuselage, derived from the Spad.

As to propulsion, the naval architect is limited by the effects of cavitation, which has appeared to be related to tip-speed and to unit loading. It is already recognized that tip-speed places an important restriction on aircraft propeller design. It has always been recognized that the "regime" of operation of water propellers is defined by definite relations of the velocity of advance, the revolutions and the diameter, which until gearing was introduced, seriously hampered the design of the prime movers. In aircraft the recognition has been retarded, although Eiffel's early reports fully indicated the importance of the V/nD factor and its significance in propeller design. This is less true in Europe, especially in England. The writer suggested gearing the Liberty Engine while the design was still on the boards at the Willard Hotel.

The use of gearing frees the engine designer and the propeller designer from hampering restrictions. This teaching, long ignored, holds many advantages now gaining belated recognition.

When it comes to cowling and cooling, the aeronautical engineers have overcome some nasty handicaps, yet the system in vogue today appears somewhat ludicrous, even giving due consideration to the cooling medium available and the form of the prime mover involved. It would be perfectly silly to advocate the setting of condensers directly in the stream adjacent to the hulls of destroyers. Independent blowers may seem equally ludicrous, but are they? The destroyer scoop method is interesting. Compressed air might serve a double purpose.

The pusher propeller has definite advantages in marine applications which might also be found to have merit in aircraft, the wake gain is no small factor. The slip stream of the tractor propeller needlessly augments the drag of whatever is in its wake, with little compensating value.

Oppositely turning propellers will eliminate torque effects and their compensation, which involves unnecessary drag. The Italians, in their racer, handled this beautifully. If adopted, the inboard turning propellers appear safest if the stabilizers are only partially in the wash, for otherwise this wash may neutralize the longitudinal dihedral effect (reference is made here to the use of laterally disposed power units, and not the tandem).

Now, to take a broad step, let us consider the reliability, continuity of service, reduced vibration and freedom from vibration of the turbine drive steam power unit, as exemplified in cruisers and great ocean liners. Admittedly in a seaway there is often vibration. Its source is not the power unit, but the location of the propellers, with their tips close to the hull, particularly when these tips come close to the surface. These conditions would rarely be approximated in aircraft, except in very rough air, where reduction in speed would, in any case, be advisable and remedial of the effect under consideration.

Let us envisage such an application to aircraft, and its advantages and disadvantages.

First an oil burning steam generator, high pressure, high temperature, delivering its superheated steam to twin power units of the high speed turbine type, suitably geared to place the propellers in their best regime. (The compound engine has certain advantages, but is not here considered because of its reciprocation being an incipient source of vibration and wear and more frequent overhaul and adjustment under the conditions of operation.) Considerable freedom exists as to the location of the power units, but it seems advantageous to have the advantage of the slip stream on the condenser scoops.

Condensers in the wing, air scoops in the slip stream to augment the flow when taking off or climbing. Destroyer scoop principle. Air discharge from the condensers carefully arranged at the top surface of the wing.

Such a power plant will lose practically no water and use but a small amount of lubricating oil. Its fuel consumption will approximate that of the best gasoline engines. Its oil consumption will be much less. Its fuel and oil will be much cheaper. The aerodynamic efficiency because of the small size of the power units and their favorable location, will be high and the resultant miles per pound of fuel quite reasonable.

The degree of vacuum will affect the turbine economy, a high vacuum is desirable from that point of view; but beyond a certain point, the condenser capacity required will increase so rapidly, that the condenser weight and resistance will absorb more power than will be gained by the increased size. A carefully considered compromise is necessary in any particular case.

The principles of such a design indicate its special suitability as a power plant at high altitude.

From studies already made in considerable detail for a 2300 HP twin turbine unit, it appears possible to realize in an initial design a weight of around 3.2#/BHP, including the power plant with its auxiliaries, the propellers and the water. The fuel plus lubricating oil consumption at cruising should be near 0.8 lb. per BHP/HR.

In larger units and with improvements in design, reasonably to be derived as the result of more exact information from the performance of the first unit, better economies of weight and

fuel consumption are to be expected. Such factors as proportions of the combustion spaces, the size and arrangement of the gas passages, the proportions of the waterwall generating and superheating surfaces; and the design of the auxiliaries to more definite conditions, are expected to contribute to these economies. Reduction in diameter of the generator tubes offers a direct saving, but increases the cost of the structure.

Offhand, these figures may appear disappointing and disadvantageous, but considered in the light of other factors which are very advantageous, even poorer figures appear attractive.

Outlined briefly, a steam unit would appear to offer the following definite advantages:

Absence of vibration, and its effects on the life of the power plant and the aircraft structure, and on the comfort of the personnel.

Absence of noise, eliminating the necessity of sound proofing and except at full power, with a proper gear ratio there should be no propeller noise to resonate with the plane.

Smooth torque, improving propeller efficiency and eliminating flutter and fatigue.

Rapid starting, full power one minute after lighting up.

Absence of ignition interference, eliminating shielding weight cost and complication.

Few controls, one throttle for each engine.

Few pilot power instruments, oil pressure, oil temperature, and tachometer for each power unit. Fuel oil cut-off.

Elimination of multiplied ignition systems, fuel systems, oil systems, mixture controls, retards.

Two (2) engines, instead of Nx(9 or 18).

Steam heat for cockpit and cabins, and galley.

Steam for ice control.

It may be possible to use skin condensers for cruising, using through wing condensers for reserve and on take off and climb, blanking the latter off for cruising.

Long life should well warrant a high cost for early units, which cost in moderate production should rapidly fall to even favorable comparison.

The Naval Architect and the Marine Engineer surely have learned things the Aeronautical Engineer will find of value.

Sphere Drag Determined by Coasting Through Still Air

R. T. SAUERWEIN, *Massachusetts Institute of Technology*

(Received June 10, 1934)

THE sphere tests described in this article constitute part of a sphere research program of Dr. R. H. Smith. It is hoped that this work will shed some light on the disagreement of existing sphere results. The coasting apparatus, described previously,[1] was used in making the measurements because a close approach to non-turbulent air conditions was possible.

The problem of obtaining a 2-foot sphere, true within close limits, having a smooth surface, and light was solved by the use of a number of

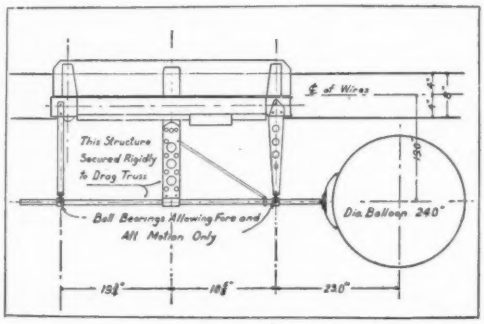

FIG. 1. Method of supporting sphere on carriage.

rubber balloons. Three or four of these were inserted one within the other, matching the thin places in one with the thick places in the others, to yield after inflation a sphere with a variation in the diameter normal to the flight path of less than a half of one percent. The internal pressure of the assembled balloons inflated to a diameter of 2 feet was about 2 pounds per square inch. Since the maximum external pressure on the balloon, due to the aerodynamic drag, was of the order of 0.002 pound per square inch, it was felt that the sphere did not deform to any appreciable extent under test conditions. To stiffen the sphere where it was attached to the spindle, a cup-shaped balsa wood collar of 10 inches diameter was bolted to the spindle and held firmly against the rear of the sphere. The assembly was mounted on a $\frac{7}{8}$-inch dural tube which, in turn, was secured rigidly to the drag truss by a suitable structure. Masts fixed to the main part of the carriage served to guide the spindle allowing only a fore and aft motion. (Figs. 1 and 2.)

The method of recording the forces is the same as has been previously described. In order that the reader may see the type of record obtained, two of the actual charts have been photographed. The chart rotates and the pointer is fixed. By comparing the two records it will be noticed that in Fig. 3 (run made in critical

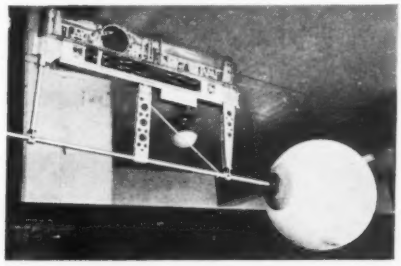

FIG. 2.

region) the force fluctuates between wide limits while in Fig. 4 (run made beyond critical region) the force remains relatively constant throughout the entire run. This characteristic difference between the drag force in the critical range and out of it was quite obvious during the tests. At speeds corresponding to the critical Reynolds number the balloon and carriage oscillated as it passed down the wires with an amplitude of about 6 to 8 inches and a frequency of about 1 per second. At all other speeds the carriage and sphere were very steady.

[1] Smith and Whipple, J. Ae. S. 1, 1 (1934).

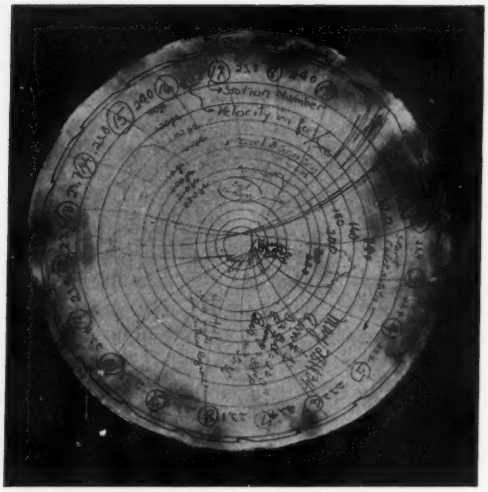

FIG. 3.

Before proceeding with the tests on the coaster it was thought advisable to mount the coaster carriage, which includes the balance, in the 7½-foot wind tunnel so that observations could be made under test conditions to be sure that the apparatus was working correctly. This procedure also made it possible to compare these results with other work made under similar conditions, to obtain the distance that the balloon must be in front of the carriage to avoid interference between the two and to obtain the tare drag of the instrument. Tests were made with an 8-inch plaster sphere and a 23.6-inch rubber balloon. The results of these tests have been plotted in Fig. 5 along with a curve obtained by Miss H. Lyon[2] using a 7.8-inch plaster sphere mounted on a spindle and supported from the top of the tunnel by two sets of Vee wires and counterweighted by a single set of Vee wires. The results obtained with the plaster and rubber spheres mounted on the coaster carriage show reasonable agreement when one considers that the smoothness of the surfaces in the two cases was not the same and that in the case of the 23.6-inch sphere

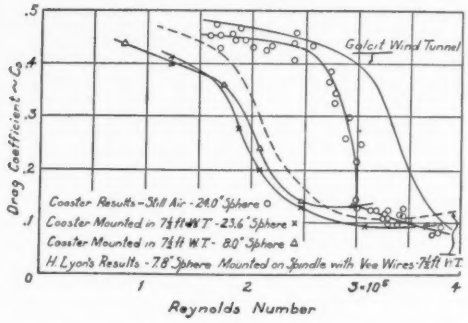

FIG. 5. Comparison of results obtained on coaster and in wind tunnel.

FIG. 4.

the speed calibration of the tunnel was probably disturbed by the size of the balloon and consequently a resulting error in the determination of the Reynolds number.

The difference between the results obtained with the two plaster spheres calls for some comment. The two spheres were not tested in the same position in the tunnel throat; the Lyon's sphere was located downstream a foot or two from the position occupied by the coaster sphere, which would place the latter nearer the honeycomb and in a position where initial turbulence was higher. This results in a lowering of the critical Reynolds number. Attention is also called to the fact that the mounting in the two cases was not the same.

The tests on the coasting wires extended over a period of two weeks during which time forty

[2] H. Lyon, R. and M. No. 1511 (1933).

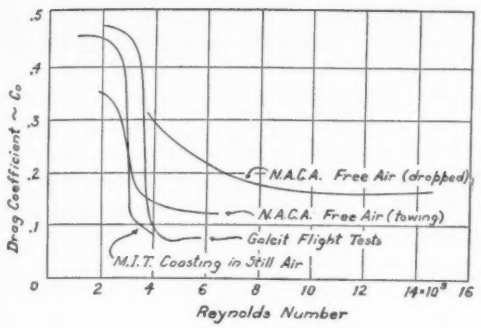

FIG. 6. Comparison of results obtained on coaster with other free air tests.

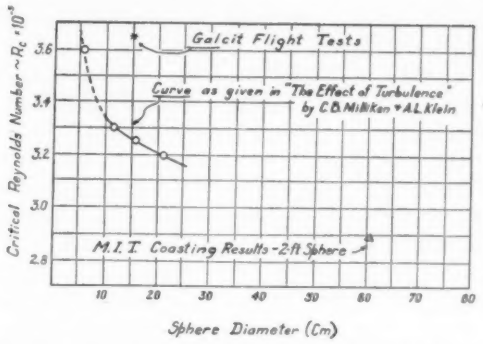

FIG. 7. Change in critical Reynolds number with sphere diameter.

points were recorded making it possible to determine a reliable curve. The results of this work are shown in Figs. 5 and 6 along with the N.A.C.A. free-air tests[3] (towing and dropped) and the Galcit flight tests.[4] The shapes of the curves obtained by the N.A.C.A. are quite different from both the Galcit and coaster curves. It is difficult to explain this difference except on the basis of the experimental difficulties and the degree of precision involved in the N.A.C.A. method.

The Galcit flight tests and the M.I.T. coasting tests yield curves that are very similar in shape but displaced with respect to each other along the Reynolds number axis. The critical Reynolds number for the Galcit results is 365,000 while for the coasting results it is 290,000. This difference might indicate that there is more turbulence in our still air experiments than in the flight tests. Referring to the Galcit wind tunnel results one sees that the critical Reynolds number is 330,000, a value lower than that in the flight tests but higher than for the coasting tests. Using the sphere as a turbulence meter one would assume in the three cases that the least amount of initial turbulence was found in the Galcit flight tests, the most in the coasting tests and the Galcit wind tunnel tests would come somewhere between these two. It does not seem reasonable that this should be true, even assuming that there is some disturbance due to heat convection in the room where the still air tests were made. Two suggestions are offered to explain these differences.

The first consideration can best be introduced by referring to Fig. 7 which shows a curve obtained by Millikan and Klein and which indicates a change in the critical Reynolds number with sphere diameter. The critical Reynolds number values determined by the Galcit flight tests and the coasting tests are also shown. An extrapolation of the given curve yields, for a 2-foot sphere, a critical Reynolds number of the same order of magnitude as that obtained by the coasting tests. There are hardly enough data at hand to draw any definite conclusions. However, the information here indicates that there is a scale effect involved which is independent of the Reynolds number. If one assumes that there is a scale effect involving a decrease in the critical Reynolds number with an increase in the sphere diameter, then one can account for the greater part of the difference between the Galcit results and the coasting results.

A second effect which also presents itself to explain these differences involves the type of mounting used to support the sphere. It will be noticed from the results in the case of the two plaster spheres of nearly the same diameter, one mounted on the coaster and the other on wires, that the wire-mounted one gave a higher critical Reynolds number than the other. It is probable that vibration of the sphere under test conditions affects the drag measurements in a similar manner to initial turbulence. If this is true, then certainly the rigidity of the type of

[3] D. L. Bacon and E. G. Reid, N. A. C. A. Report No. 185 (1924).

[4] C. B. Millikan and A. L. Klein, Aircraft Eng., London, Aug. (1933).

mounting used can affect the critical Reynolds number obtained.

Further tests should be run over as large a range of Reynolds numbers as possible and under varying conditions of initial turbulence, including still air or flight test conditions, to determine whether or not there is a scale effect independent of the Reynolds number and if so to ascertain its magnitude. Tests should also be made of the type of mounting, to determine what effect the rigidity of the method of supporting the sphere has upon the value of the critical Reynolds number obtained. It will finally be of interest to determine the effect of vibration on separation of the boundary layer and on the flow pattern, first on spheres, which should be sensitive indicators of such an effect, and then on wings and other forms.

Books Received

The Economics of Air Mail Transportation, by PAUL T. T. DAVID; The Brookings Institution, Washington, 1934; 235 pages; $2.00.

This study of the Air Mail was written before the recent investigation and cancellation of Air Mail contracts, but was revised to take account of recent developments. It traces the history of the Air Mail service from its beginnings under government operation. Air Mail legislation is given a critical examination and the contractual relations of the government with transport companies are reviewed. A chapter is devoted to the various forms of regulation which have been imposed on the industry. Charts giving the statistical progress of Air Mail transportation make the book a valuable reference work. The conclusions reached by the author are in the field of controversy, and his program for the future gives his suggestions for the present emergency as well as for the future.

The Air Menace and the Answer, by ELVIRA H. FRADKIN; The Macmillan Company, New York, 1934; 331 pages; $3.00.

Mrs. Fradkin has collected in this book a great deal of current information concerning chemical warfare and its relation to aerial warfare. She is a proponent of the abolition of, not only aerial warfare, but all war, and presents her arguments with the zeal of those who champion disarmament and pacifism.

The Australian Flying Corps in the War of 1914-18, by F. M. CUTLACK; Angus & Robertson, Ltd., Sydney, Australia, 1933; 485 pages.

This book is the authoritative history of the Australian pilots during the World War. It has been compiled from official diaries and sources. It follows the activities of the Corps from Mesopotamia to France. It is well illustrated and documented.

Skin Friction Papers.—The Library of Congress has recently issued a 46-page mimeographed pamphlet on Skin Friction and Boundary Flow, covering the subject as far as December 31, 1932, and adding some more recent papers. Its main theme is the determination, by theory or experiment, of the velocity distribution in the boundary layer, and the attendant skin friction, of steady flowing slightly viscous fluids such as air and water. It was compiled by Dr. A. F. Zahm, who occupies the Daniel C. Guggenheim Chair of Aeronautics in the Library, and Mr. C. A. Ross, member of the aeronautics staff.

Superaerodynamics.—In the *Journal of the Franklin Institute* for February, 1934, Dr. A. F. Zahm has an article on superaerodynamics, a term adapted from General Crocco's "superaviation." After summarizing Newton's doctrine of resistance in a stream of discrete particles it gives some extensions of that doctrine, comparing the pressure and resistance in such a stream with those found in normal air. In conclusion he recommends for academic research the dynamic pressures in tenuous air with those computed for Newton's hypothetical fluid.

A Reflection Plate Representing the Ground

E. A. STALKER, *University of Michigan*

(Received May 2, 1934)

IN testing vehicles for air resistance on or near the ground the simulation of actual conditions requires that there be a surface at which the normal velocity be zero. In reality a vehicle moves at the same rate relative to the roadbed and the air if there is assumed to be no natural wind, while in a wind tunnel the vehicle and roadbed are stationary relative to each other.

Several methods have been proposed to represent the roadbed. Eiffel, I believe, used belts moving with the linear velocity of the wind tunnel stream; other experimenters have given reality to the reflection principle so useful in theoretical physics by using two models, one inverted with respect to the other.

The use of belts requires a rather complicated mechanism which becomes particularly troublesome at high speeds due to the natural tendency to vibrate and shift position.

The reflection principle assumes in the case of aerodynamics that the flow is streamline because only then can there be assurance that the vertical velocity induced in the stream by the model and its image are precisely equal and opposed. In fact, if the flow is turbulent and of a certain uniformity even initially, it cannot remain so and a surface of hills and valleys, usually of unknown contour, is presented.

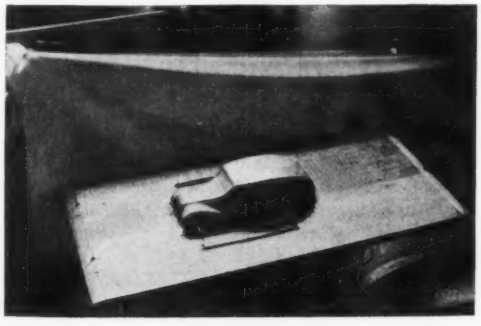

FIG. 1.

In some fairly recent tests on automobile bodies at the University of Michigan the writer suggested the use of a plate upon which the boundary layer would be removed or energized so that it did not present a drag to the stream. The first attempt, as shown in Fig. 1, embraced the use of suction to remove the boundary layer. Measurements were made on the car resistance by this method, by the reflection method, and with no roadbed of any sort. Full size tests are in progress which will eventually show the relative merits of these modes of experiment.

It is to be said that removing the boundary layer by suction causes a small downward component which in many tests is objectionable. To suppress this disadvantage modification of the plate has been recently undertaken. It is being covered with overlapping metal sheets for the ejection of air as well as its withdrawal. See Fig. 2. If the direction of flow is alternated the boundary layer will be annihilated and the vertical components will be zero to a high degree of truth since the slots are quite close (2 to 3 inches) together.

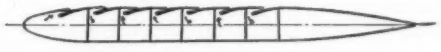

FIG. 2.

In the case of a plane surface the adjustment of the jet velocities does not become too complex once the flow is properly distributed in each compartment. The scheme is susceptible to adaption to surfaces of curved contour but probably would become too complex for any but simple curves.

Assuming that the boundary layer is situated between the plate and the locus of a velocity 97.5 percent of that 6 inches above the plate, Fries and Lowry[1] found the layer at the model

[1] A. H. Fries and R. D. Lowry, *The Air Resistance of Automobile Bodies.* Unpublished Thesis, University of Michigan, 1931.

location to be forty-five sixty-fourths of an inch thick. When the layer was reduced to twenty-seven sixty-fourths the drag of the model was increased 17.6 percent. Using the reflection plate with approximately this degree of boundary layer removal as a standard, they found that the free model showed 11.0 percent more, with the reflection method 17.2 percent more and with the flat plate without boundary layer removal 13.2 percent less drag.

The device presents a rational solution to the roadbed problem and should find use particularly where empirical data, indicating that other devices give satisfactory results within the order of accuracy sought, are not available.

Meetings of the I. Ae. S.

THE first Technical Meeting of the Institute was held in the R. C. A. Building, New York City, on Thursday, March 29th, at 5:30 P.M. Mr. Elmer A. Sperry, Jr., spoke on "The Development of Blind Flying." He traced instrument flying from its beginnings and showed a picture of his brother, Lawrence Sperry, who is regarded as the pioneer blind flying pilot. Slides which illustrated the development of blind flying instruments were shown and explained.

Mr. W. Lawrence Le Page read a paper on "Rotative Wing Aircraft" at a Technical Meeting of the Institute held on Thursday, April 26th. He traced the history of the autogiro, the cyclogiro and the gyroplane and discussed their technical possibilities. The talk was followed by a lively discussion by Edward P. Warner, E. Burke Wilford, Charles L. Lawrance, Agnew E. Larson, Elliott Daland, Paul Hovgaard, Captain F. T. Courtney and Mr. J. D. North, an English member of the Institute.

Technical Meetings will be held monthly or oftener. After the meetings, the members dine together to discuss further the subject of the meeting.

Weekly luncheons have been held on Wednesdays by members of the Institute at a restaurant near the Skyport. These luncheons are informal and members are finding them interesting, as they bring together many old and new friends who discuss freely aeronautical matters of current interest. The luncheons have been discontinued for the summer.

A "Stormoguide" has been given to the Skyport by the Taylor Instrument Companies of Rochester, New York.

Visit to U.S.S. Lexington

About two hundred members of the Institute and their guests visited the aircraft carrier *Lexington* on Thursday, June 7th. Rear Admiral E. J. King requested Rear Admiral John Halligan, Commander Aircraft of the U. S. Fleet, to extend a group invitation to Institute members. Admiral Halligan and his staff received the party and officers escorted the members in small groups over the ship.

Election of Fellows

The following members of the Institute have been elected to the grade of Fellow. The election of Fellows of the Institute is now limited to ten each year. All Fellows are proposed and elected by the Fellows themselves. Nine more may be elected during 1934.

Charles P. Burgess	Charles A. Lindbergh
Frank W. Caldwell	Grover Loening
V. E. Clark	George J. Mead
James H. Doolittle	C. B. Millikan
W. F. Durand	Alfred S. Niles
James H. Kimball	H. J. E. Reid
W. B. Klemperer	Starr Truscott
Emory S. Land	Edward P. Warner
Charles L. Lawrance	T. P. Wright

Necrology

Fred M. Harpham, an Industrial Member of the Institute, died at Akron, Ohio, on June 12, 1934. He was Vice-President of the Goodyear Tire and Rubber Company and the Goodyear-Zeppelin Corporation. He was 59 years old. Mr. Harpham was widely known as an advocate of airship development and transoceanic airship lines. He was one of the executives most actively associated with the construction of the airships Akron and Macon.

Lieut. Henry B. Harris, a Pilot Member of the Institute, died on June 15, 1934, at Elmira, New York, as the result of an accident in an automobile which was towing a glider. Lieut. Harris was born October 23, 1907, at Darien, Conn. He was pilot in charge of meteorological research flying at the Massachusetts Institute of Technology. He was an expert instrument pilot and had made flights almost daily since 1932 as part of the investigation being made at M. I. T. of the meteorological, bacterial, and chemical variations in the atmosphere at altitudes up to approximately 18,000 feet.

Letters to the Editor

Speed of Air-Driven Rotors in Gyroscopic Instruments

Some difficulty has been experienced by readers of my article in the January issue in checking the variation of the windage factor with Reynolds number which I have shown for the Artificial Horizon. Such difficulty is eliminated if it is remembered that the differential pressure Δp, referred to in the formulae, is the drop occurring across the nozzle alone and that this is less than the total drop across the instrument.

Part of the pressure drop across the instrument occurs in the passages leading to the nozzle and is not effective on the nozzle velocity. It was found experimentally that for the Artificial Horizon Δp was approximately 45 percent of the total drop across the instrument over a wide range of suction values, there being serious obstructions in the passage of the air before it reaches the nozzle. In the case of the Directional Gyro the air flow is comparatively free from obstructions and Δp is taken to be the total drop across the instrument.

ATHELSTAN F. SPILHAUS
Massachusetts Institute of Technology,
May 8, 1934.

Extension of Weather Observations by Means of Airplanes

Members of the Institute will be interested to know that, through cooperation of the War and Navy Departments, the Weather Bureau's program of upper air observations by means of airplanes will be materially extended after July 1, 1934. Daily flights, beginning about 5.30 A.M., E.S.T., will be made at about twenty stations, including six to be operated by the Weather Bureau and seven each by the War and Navy Departments. The stations are fairly well distributed over the country. The flights, which will as a rule extend to a height of fifteen to twenty thousand feet, will provide information concerning temperature, humidity and other conditions by means of an instrument known as a meteorograph. Upon the return of a plane to the ground, the record will be at once computed and the data will be transmitted over the teletype system of the Department of Commerce to numerous stations throughout the United States for use by the officials of the Weather Bureau engaged in forecasting, and for the information of others interested.

In addition to the twenty stations which will make flights on a regular schedule, there will be occasional flights at Boston, through cooperation with the Massachusetts Institute of Technology, and at about half a dozen naval air stations where facilities are not yet available for making these flights except during the daytime. Finally, it is expected that airplane flights will be made from about seven naval ships, but these, likewise, will be at irregular hours.

This extension of observations by means of airplanes marks the first definite step in carrying out the recommendations of the Science Advisory Board. In many respects it is the most important of those recommendations, since the effective use of air mass analysis methods in forecasting requires above all else detailed data in the upper air. The records from these twenty stations, supplemented by occasional reports from other points, and by upper wind observations by means of pilot balloons at a large number of stations well distributed over the country, will provide a basis for studies of the structure of the air, which constitute a necessary preliminary to the actual adoption and use of air mass analysis in day to day forecasting. As such, this step is regarded as an extremely important one.

W. R. GREGG
U. S. Weather Bureau,
May 7, 1934.

Strength of Sheets

The difficulties of analysis of monocoque construction in aircraft have resulted in numerous tests being made to determine the allowable loads on flat and curved sheets, with and without stiffeners. In the empirical equations derived from these, the strength of the stiffener may be calculated by one of the various column formulae, whose results are in close agreement with test results. In analyzing the sheet, however, no satisfactory theoretical formulae have yet been developed.

It has been noted that recent studies made on the strength of sheets under this type of loading have been based upon fundamental elasticity equations, in the solution of which arbitrary assumptions have been made as to stress distribution.

To procure less arbitrary stress distribution data the writer is conducting tests on celluloid sheets subjected to compressive loadings. The apparatus in use consists of two V grooves between which the sheet fits and which simulate the stiffeners in aircraft construction. The load is applied to the ends of the sheet by heavy steel plates. The sheets will be examined photoelastically and strain readings will be made. The length to width ratio will be studied. The type of fundamental elasticity equation and its solution will be determined by an analysis of the stress distribution and deflections observed in the sheet.

AUGUST ZINSSER, JR.
Massachusetts Institute of Technology,
May 15, 1934.

INSTITUTE OF THE AERONAUTICAL SCIENCES

AIMS

The Institute of the Aeronautical Sciences was organized to promote the application of science in the development of aircraft. By grading members according to their qualifications, recognition is given to outstanding professional work and an incentive is provided younger members for the investigation of new aeronautical problems. The presentation and discussion of technical papers at Institute meetings afford members opportunities to hear and meet the leaders in specialized fields of the aeronautical sciences. Through the publication of the Journal of the Aeronautical Sciences, members are provided with an opportunity to publish and to read new knowledge and new applications in the field of aeronautical endeavor. By including in its membership the leading aeronautical specialists in other countries an interchange of international aeronautical scientific progress is made possible. The organizers of the Institute believe that the future development of aircraft depends on scientific research and engineering experiment applied to design. As practically every scientist and engineer already belongs to his own special professional society, in order to bring together all specialists contributing to aeronautical progress, the dues of the Institute have been made nominal. Through the foregoing aims, as well as others too obvious to specify, the Institute is providing the United States with a scientific society similar to those which have had such a beneficial influence on aeronautical progress in other countries.

THE SKYPORT, *the headquarters of the Institute, is located on the fifty-fourth floor of the largest office building in the world. It provides members with a convenient and attractive meeting place and reading room.*

DUES

The entrance fee for all members except Honorary Fellows, Honorary Members, Benefactors and Technical Members, is five dollars and that for Technical Members three dollars. The annual dues are: Scientific Members, $5.00; Engineering Members, $5.00; Pilot Members, $5.00; Industrial Members, $5.00; Technical Members, $3.00.

APPLICATION for MEMBERSHIP

Application forms for membership may be secured from any member of the Institute or by requesting them from the Secretary.

Applicants are required to furnish the names of several persons, preferably members, who are acquainted with their professional work. After these references have been communicated with by the Institute, the Admissions committee recommends the grade of membership. This recommendation is then passed on by vote of the Council after which an invitation to join may be extended to the applicant.

INSTITUTE OF THE AERONAUTICAL SCIENCES

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